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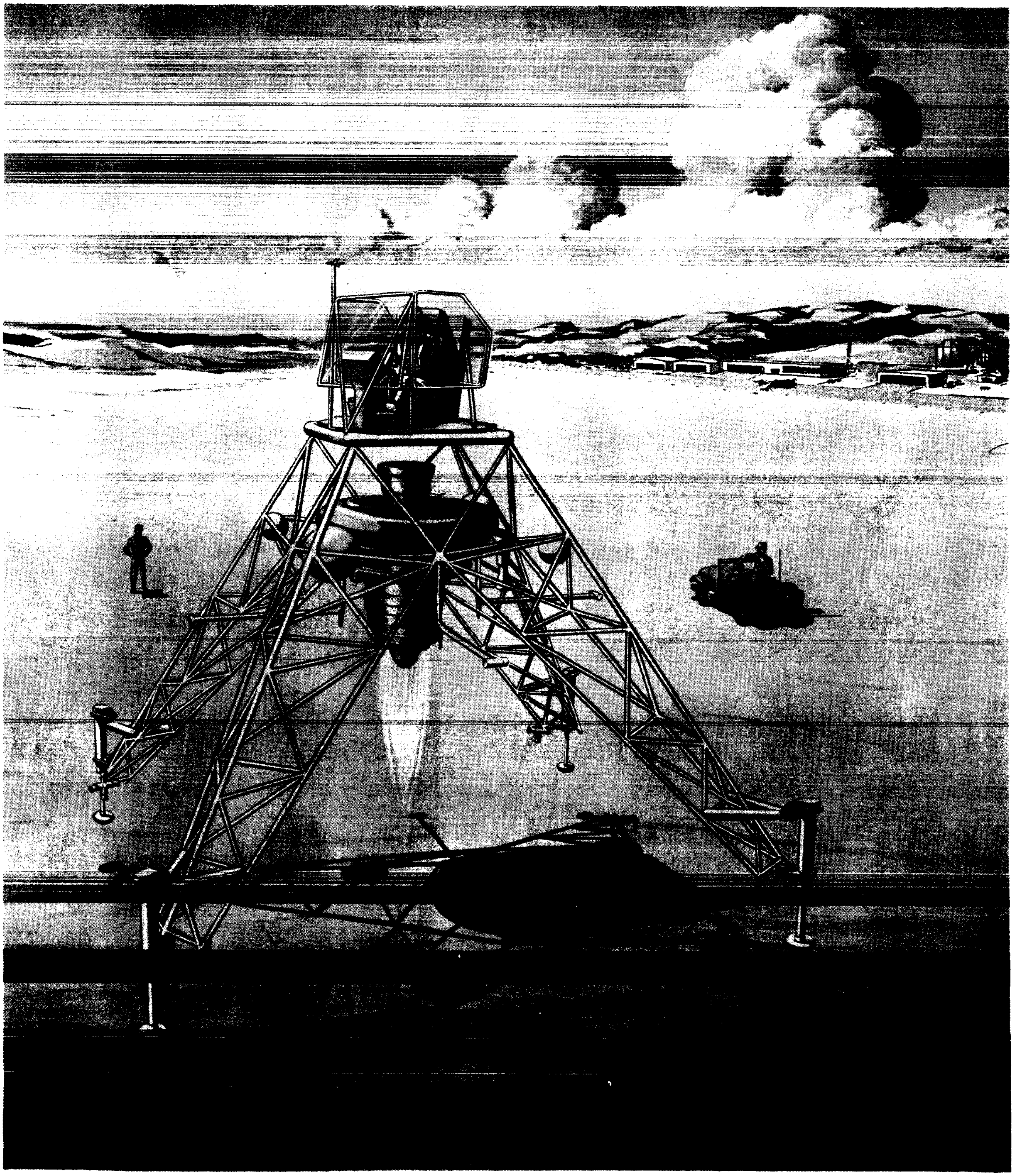
FEASIBILITY STUDY FOR A LUNAR LANDING FLIGHT RESEARCH VEHICLE

REPORT NO. 7161-950001

MARCH 8, 1962



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ABSTRACT

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This report presents the results of a study conducted by Bell Aerosystems Company during January and February 1962, under Contract NAS 4-174. The purpose of the study was to establish the characteristics and conduct a preliminary design of a free flight lunar landing research test vehicle. A study of basic lift methods indicates that a vehicle using a combination of one turbofan jet engine and hydrogen peroxide rockets can provide a reasonable simulation of a manned lunar landing and can be produced at a minimum of time and cost.

A preliminary design of a one-man research vehicle is presented, including a description and analysis of the open truss type structure and the propulsion, flight control, cockpit, and electrical systems. A vehicle stability and performance analysis is included. A study of reliability, safety, support equipment and facility requirements, wind tunnel requirements, schedule, cost, and future growth is presented.

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I. INTRODUCTION

The lunar exploration program which is currently being implemented will encounter new and unusual environments which are unfamiliar and foreign to our presently developed knowledge of flight operations. Not only are new engineering design concepts required to meet the challenge, but in addition, for many phases of the Apollo flight mission, the basic specifications and requirements on which these new engineering designs will be based must be reviewed, re-evaluated and new requirements appropriate to Apollo established. Typical of these mission phases is the lunar approach, hover and touchdown.

For conventional air breathing aircraft and helicopter, the entire history of aircraft design and flight testing has been used to develop basic engineering specifications for such items as low speed handling characteristics, landing gear load factor and stability requirements, aircraft safety performance limitations, and basic piloting procedures to which engineering designs must conform. Even for earth based VTOL aircraft, much recent work has been accomplished both analytically and in free flight research, to develop similar design requirements and specifications for control characteristics and performance and safety during the takeoff and landing phases of flight of these hovering aircraft.

For lunar landing, however, no such previous effort exists and a program of research and testing is urgently needed to develop the necessary criteria and requirements on which the Apollo design can be based. In the history of manned vehicles, there are few cases in which the design of an operational system has been committed without the benefits of prior investigation and flight research with experimental vehicles. It is to this need for flight test research and for a tool from which basic Apollo requirements and specifications for lunar landing can be determined, that the study of a Free Flight Lunar Landing Research Test Vehicle has been devoted. The basic objective has been to establish the design concept of a test vehicle which can be constructed quickly and at low cost, and which will provide a realistic simulation on earth of landing operations in a lunar environment. Such a vehicle can be used not only to develop necessary engineering specifications and requirements, but can also aid in evaluation of early Apollo engineering concepts, and help to develop requirements for flight training and flight procedures for the Apollo vehicle.

Two characteristics of the lunar environment are basic to the design of a free flight vehicle which can simulate lunar landing trojectories on earth. First, the low lunar gravity, which is about 1/6 that of the earth, and second, the lunar atmospheric pressure, which is about 10^{-13} mm of mercury and causes no appreciable aerodynamic forces on the lunar landing vehicle. For realistic earth simulation, a free flight test vehicle must provide first a means of reducing the effects of earth gravity while preserving similarity of vehicle mass and moment of inertia, and second, a means of balancing aerodynamic forces so that the vehicle appears to operate in a vacuum environment. In providing this simulation, the test vehicle should not by itself introduce any unrealistic side effects.

The importance of these two factors can be seen by comparing a horizontal translation maneuver on the moon and on the earth. Figure I-1A shows the horizontal velocity versus time for a lunar landing vehicle in a representative maneuver in approaching a landing site. Translation is accomplished by tilting the vehicle so that the horizontal component of the lift rocket can be used to accelerate the vehicle laterally. The solid line if Figure I-1B indicates the pitch attitude required. The vehicle would pitch nose down foreward, to obtain forward velocity. When sufficient foreward velocity was attained, the vehicle would return to a zero pitch angle. As the destination is approached, the vehicle would tilt nose up backward, in order to bring the vehicle to a stop.

The dotted line of Figure I-1B shows how the same velocity time profile would be accomplished by an earth-based VTOL vehicle with fixed engines. Since the lift vector on earth is six times that on the moon, whereas the horizontal component required for acceleration is the same on earth as on the moon, the earth vehicle will tilt only one sixth as much. As speed builds up, the tilt angle must be increased to compensate for drag. When the desired velocity is reached, the tilt angle is decreased and held at the attitude required to overcome drag. As the destination is approached, the vehicle must be tilted nose upward to slow the vehicle down and the tilt reduced gradually to zero as velocity and drag decrease.

It can be seen that the earth VTOL has duplicated the velocity/time profile of the lunar vehicle but the pitch attitude maneuver required on earth is so different from that required on the moon as to render the earth vehicle useless for simulating attitude control. For a successful simulation, five-sixths of the earth weight and the drag force must be counteracted by a suitable thrust vector.

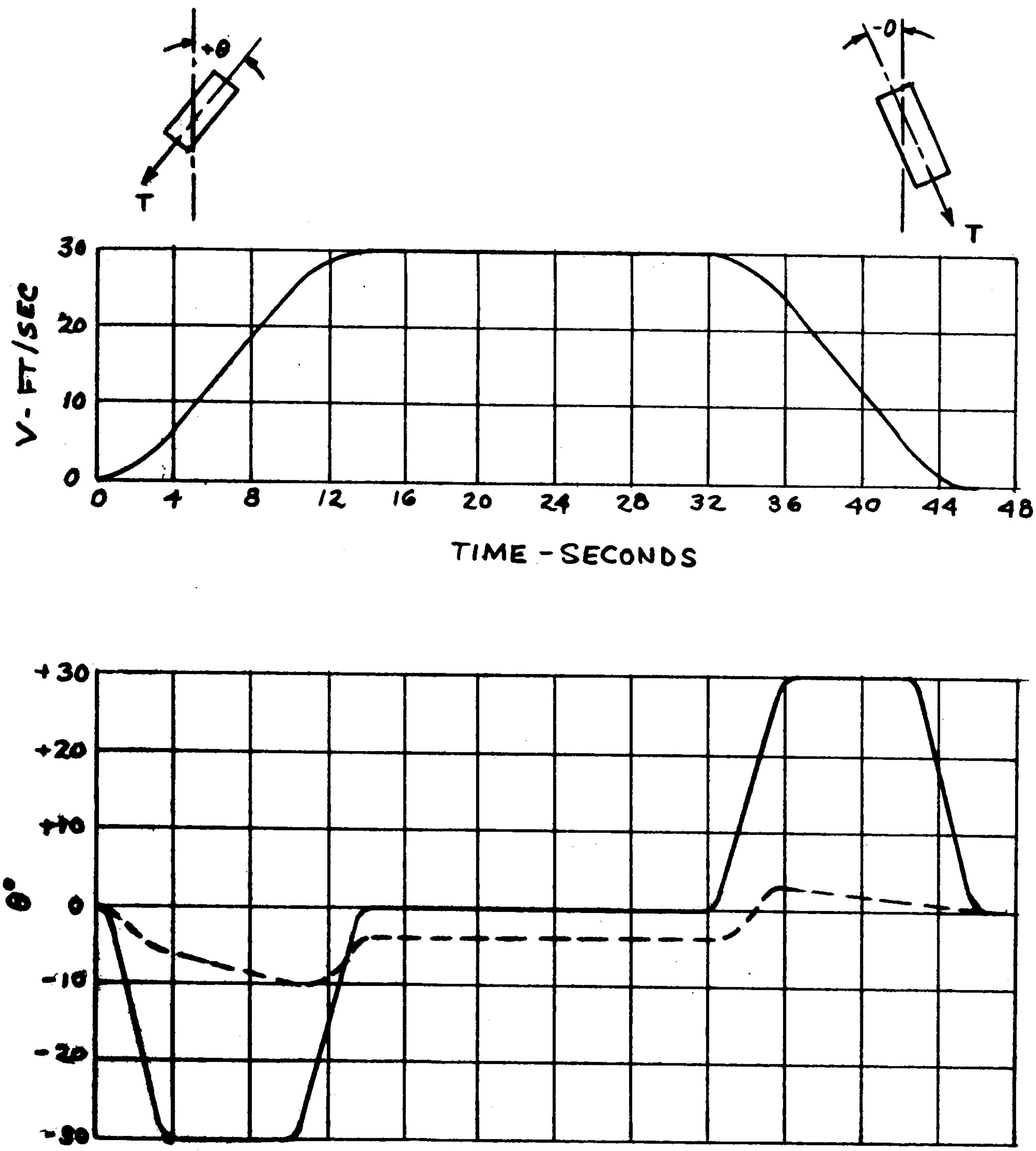


Figure I-1. Velocity and Pitch Angle During Translation

This can be accomplished only by designing the research test vehicle in a manner so that the attitude of the thrust vector which balances earth gravity and aerodynamic drag can be adjusted independently of the attitude and motion of the basic vehicle itself. The response of the vehicle to attitude controls and rocket lift engines is then preserved and realistic flight situations can be simulated.

A. PROBLEMS REQUIRING RESEARCH BY SIMULATION

The lunar landing research test vehicle which is presented in this study will provide an essential test tool with the necessary flexibility to allow flight investigation in many areas. Some of the basic problems of lunar landing for which this vehicle can provide needed data, are outlined below.

1. Control With No Aerodynamic Dumping

The Apollo Vehicle will involve for the first time a pilot controlled landing operation in which no aerodynamic forces will exist. While some earth based vehicle designs have encountered low aerodynamic damping of angular motion, no vehicle has yet encountered the complete lack of aerodynamic damping of both angular and linear motions. As such, current specification for control characteristics and handling which has been accepted by pilots do not apply to lunar landing. Exploratory research on these requirements can be accomplished by electronic analogue simulation, however, verification and acceptance of fixed based simulator results can be accomplished by free flight research of piloted vehicles.

2. Effects of Low Lunar Gravity

Similar to (1) above, specifications have been developed for earth based VTOL aircraft defining required handling characteristics, thrust modulation, thrust margins, etc. for hovering operations in an earth gravity environment. New definitions are needed for operation in lunar gravity where hovering thrust to mass ratio is considerably different, and accompanying rate of climb, rate of sink, and translation capability are significantly changed. As above, basic exploration fixed based simulators can only be evaluated and confirmed by free flight research.

3. Pilot Position and Visibility

Since the Apollo vehicle configuration and pilot position may not represent the most ideal for vehicle control, an investigation is required into the extent to which the position of the pilot and the restraints imposed upon him from the pressure suit and from the seat will affect his capability to control the vehicle. If the pilot seating is such that the pilot sits upright with the vehicle in normal flight attitude, during the landing phase he may be flying on his back with his feet up in the air. Experiments with the X-13 "Tail Sitter" VTOL indicate that if the pilot's seat were rotated forward 45°, the pilot was in a position whereby he could exercise reasonably good control, however, space limitations of a lunar vehicle may not permit rotation of the seat, thus requiring the pilot to control the vehicle from a reclining position. Not only will the pilot be controlling from this position, he will also be encumbered with a pressure suit which will further restrict his capability for making precise control movements. Finally, there may also be some cross-coupling from kinesthetic and vestibular cues resulting from the necessity of controlling the vehicle from an atypical body position. Prior to launching a lunar mission, these problem areas should be investigated in a realistic environment to determine their effects and subsequently, efforts undertaken to either design around them or to train the pilots to a sufficiently high degree that they are able to cope with these problems.

The investigation of viewing techniques on controllability should take into account the visual environment in which the vehicle will be operated. This includes natural as well as induced characteristics of the environmental field. The natural characteristics include brightness, glare, contrast, texture and color. The induced environment includes changes in the visual environment resulting from the presence of the vehicle, such as glowing dust and other impingement effects. With these characteristics in mind, a thorough investigation of both direct and indirect viewing techniques can be accomplished. Investigation of direct viewing techniques should include configuration, i.e., size, shape, and position of windows as well as optical anomalies, i.e., position and motion distortion and the optical characteristics of the window. The indirect viewing techniques include periscopes, closed loop television, radar, infrared techniques and optical systems. The variables of interest include the size of the visual field, the number of degrees that can be scanned, the viewer characteristics, location of viewer and controls with respect to the pilot, resolution, signal to noise ratio and distortion. Based upon the results of these studies, it will be possible to determine the viewing techniques and design requirements that will permit the pilot to execute a lunar landing with the minimum possibility of error.

4. Control System Characteristics

There are a number of control problem areas that need further study to insure a safe lunar landing -- these include the determination of the most effective type of control system, that is a control system that utilizes acceleration commands, rate commands or position commands to control vehicle attitude. Within each of these types of control systems, a better understanding of the control sensitivity and control magnitudes required is needed. Consideration must also be given to fail-safe requirements for the control system. Based upon the results of studies on the required control characteristics of a lunar vehicle, a determination of the optimum pilot control configuration, and evaluation of this control configuration under conditions that approximate the actual lunar landing conditions should also be made.

5. Determination of Pilot Landing Display Parameters and Their Instrumentation.

Since the lunar landing represents a rather unique vehicle control situation, a thorough investigation of the pilot landing display parameters and instrumentation should be undertaken. This should include determination of what information should be displayed, how the information should be displayed, the requirement for quickened displays and/or predictor type displays. As a minimum, it would appear that the pilot should be provided with precise attitude information, rate of descent, velocity, thrust, absolute attitude and flight path information.

6. Landing Tipover Problems Due to Low Gravity.

The precision required for control of the vehicle at touchdown will be much greater for a lunar landing than for the landing of a comparable vehicle on earth. This results from the large increase in tipping tendencies due to the change in relationship between vehicle inertia and static restoring moment due to gravity. If, for example, the maximum safe lateral touchdown velocity on earth were 5 feet per second, for a similar system on the moon, it would be reduced to 2 feet per second. Thus for control of velocity, the requirements are approximately 2-1/2 times as stringent on the moon as on the earth. This raises serious question as to the most effective means of presenting information to the pilot that will enable him to exercise this precise control. Prior to finalization of the display system in the lunar vehicle, this whole problem area should be investigated under as realistic conditions as possible. As the control and display systems of the lunar vehicle are evolved, studies should be undertaken to develop control techniques for hover, descent and landing. These should include various techniques of controlling trajectories, forward and lateral translations and vertical descent so as to minimize fuel requirements, pilot workload and time required to accomplish the landing.

Also, the requirements for stability on the design of the lunar landing gear should be given serious attention. Design specifications are well documented for aircraft and helicopters in terms such as safe tipover angles and allowable vertical and lateral velocities. These are defined, however, on the basis of flight experience gained under earth gravity conditions and need re-evaluation for lunar operations.

7. Control of Vehicle Translation

Horizontal translation of a rocket supported vehicle can be accomplished by tilting the entire vehicle and its rocket engines to provide a horizontal component of thrust. Translation may also be accomplished by using horizontally mounted auxiliary thrusters. It can be shown that tilting of the entire vehicle is more economical in use of propellants. However, for hovering on the moon the vertical thrust vector is only one-sixth of that required on earth, whereas the horizontal thrust vector needed for lateral acceleration is the same as that required on earth, so that vehicle tilt angle can be very large. For example, to translate a thousand feet and come to a stop within thirty seconds requires a tilt angle of 40°. This raises questions of man's ability to stabilize vehicle attitude while in a 40° tilt and complicates the problem of providing good visibility of the terrain below and around him. Thus, additional research is required to establish which is the preferable method for a specific mission.

8. Control of Lift Rockets

Vertical thrust vector magnitude must be controlled precisely in order to accomplish vehicle touchdown at essentially zero vertical velocity. However, to provide the highest reliability, it may be desirable to avoid throttleable rocket engines. Vertical thrust vector could be controlled by pulsing one of the lift rockets on and off, or by tilting opposing pairs of rockets outward. Again, since manual control factors are involved, an optimum solution will require additional research.

9. Control of Attitude Rockets

Vehicle attitude control can be provided by gimballing the main lift rockets, or by use of auxiliary reaction control rockets. Both have been used successfully. The choice depends not only on weight, complexity, and reliability, but also on moments required for optimum manual control, to be determined by flight research.

10. Other Research Areas

In addition to the problems outlined in the foregoing sections, a number of other areas should be investigated with the lunar landing simulator. These include:

a. Verification of Apollo subsystem design and techniques.

b. Training and performance/proficiency assessment.

c. Personnel selection criteria development.

d. Optimization of the final portion of the lunar let-down trojectory with manual flight control.

B. SPECTRUM OF SIMULATORS AND PLACE OF FREE FLIGHT SIMULATION

Flight simulation is a basic tool which has been developed to provide at low cost and in short time, design data and training which could otherwise only be obtained by flight of the actual full scale system. Simulation covers a broad range of types with the individual methods of simulation intended to accomplish a specific objective. The various schemes of simulation for flight research can be classed as:

1. Electronic
2. Fixed Base Mechanical
3. Free Flight Test Vehicles
4. Free Flight of Operational Vehicles under simulated conditions.

These methods are listed in order of approaching realism to operational flight conditions.

The most realistic simulation is of course accomplished with the actual operational flight system. Unfortunately, this requires committing design, development and fabrication of the operational system without the ability to explore design requirements and design concepts.

Electronic simulation allows exploratory testing without committing hardware of any kind, can be accomplished rapidly and allows a wide range of simulation at low cost. Unfortunately and particularly when the research involves pilot operated systems, the important element of actual flight environmental and stress conditions is lacking in electronic simulation. For earth based flight vehicles, considerable experience has been developed in correlating electronic simulation results to free flight tests, so that this method has become extremely valuable as a design tool. For lunar flight however, this correlation has not been developed, so that the need for flight research data to support and complement electronic simulation is urgent.

An intermediate step between electronic and flight simulation is accomplished with the fixed base simulators, such as mechanical centrifuges and gimbal and tether systems which introduce the next step of allowing actual physical motion. In these systems some of the physical stresses induced on the pilot are introduced. Fixed base mechanical simulators are still limited, however, to reproducing only a portion of the environment of free flight and for reasonable limitations in size cannot duplicate the complete freedom of free flight, or the stress levels which are a part of actual flight. In some cases, fixed base simulators can, while duplicating certain real flight conditions, introduce other conditions which are not characteristic of actual flight, such as the effects of coming against mechanical stops.

The free flight research vehicle is the only method which bridges ground based testing and flight testing of actual operational vehicles. The free flight research vehicle provides therefore, a complement and extension to ground testing.

II. INVESTIGATION OF BASIC LIFT METHODS

In order to select a means of auxiliary lift required for an earth-based free flight vehicle which can simulate the dynamic response which will be experienced in a lunar gravitational environment, several lift methods were considered and compared.

A. Helicopter lift by direct gimbal coupling to the vehicle
B. Helicopter lift by gimbal-and-tether coupling
C. The thrust of a vertically-oriented fan
D. Liquid-fueled rocket engine
E. Turbojet engine

The factors considered in the judgment of each means of auxiliary lift were configuration, relative ease of eliminating aerodynamic effects, weight and performance, fidelity of simulation and cost, availability and scheduling requirements.

A. HELICOPTER LIFT BY DIRECT GIMBAL COUPLING

The use of a helicopter as a means of auxiliary lift for the lunar landing simulator requires the vehicle to be gimbal-coupled with two degrees of freedom to a supporting structure beneath the helicopter. This coupling would require the addition of extended landing struts to the helicopter to avoid ground contact with the vehicle during touchdown. The combined weight of the vehicle and the added helicopter structure would required the load-carrying capacity of the helicopter to be in the range of 2000 to 2500 pounds. An existing helicopter would require fairly extensive airframe modification to sustain this mode of loading.

A helicopter used as the means of auxiliary lift must maintain 5/6 of the weight of the capsule during all conditions of vehicle maneuvering and, in addition, provide the proper level of thrust tilt to overcome combined system drag. Helicopter thrust and attitude controls are required consistent with these requirements. The helicopter thrust and attitude command system must have reference data on either, (1) wind velocity and direction in order to determine and correct for aerodynamic forces and moments, or (2) three-axis acceleration command data as a function of vehicle attitude and thrust conditions. This information would be compared with actual accelerations to provide helicopter control inputs. Because of the inherent difficulties in obtaining velocity measurements in diverse directions, which in turn must be related to throttle control and attitude control, the preferred method, based upon our studies, is to use acceleration reference commands (as functions of vehicle thrust and attitude conditions) as the controlled quantity. However, the helicopter is highly sensitive to aerodynamic effects caused by gusts, winds and the magnitude and direction of flight velocity due to its large disc diameter and low disc loading. A fast-response control system would be required to maintain precise levels of thrust and attitude. It is felt that the response rate of the helicopter's attitude control system would be a serious limitation in this respect. The fidelity of simulation is of course directly related to the degree to which disturbances to the capsule system are minimized. It is the helicopter's inherent sensitivity to aerodynamic effects which represents a potentially serious drawback.

It is from these considerations of weight, requirements for helicopter redesign, and low fidelity of simulation that lead to the conclusion that the helicopter, while feasible, is a more costly solution to the problem.

B. HELICOPTER WITH TETHER-SUPPORTED VEHICLE

It is possible to use a helicopter without modification by hanging a lunar landing simulator within a large gimbal ring, by a long cable or tether beneath the helicopter. This method of support for a free flight lunar landing simulator has the advantage of not requiring major structural rework or redesign of the helicopter. However, the problems of producing a valid simulation are extensive. As in the previous case, the helicopter must maintain a steady lift equal to 5/6 of the weight of the vehicle under all conditions of vehicle thrust and maneuvering, which involve the same thrust stabilization problems as for the previous case. However, the problem of drag compensation appears to be even more stringent because of the unique position-relationship required between vehicle and helicopter. In order to provide drag compensation, a definite angular relationship must exist between helicopter and vehicle as a function of forward speed to provide the force necessary to overcome vehicle drag. Even neglecting short period disturbances, the method of providing good control of position relationship appears very difficult. Although this scheme is technically feasible, it presents a considerable development problem, since the tether cable-angle measurements would require special developments. Fidelity of simulation, therefore, appears to be one major drawback of this approach. The vehicle weight would be somewhat greater than the previous case because of its landing requirements.

C. DUCTED FAN LIFT METHOD

Detailed performance estimates have not been made on a specific ducted fan approach. However, ducted fan possibilities were extrapolated from a design made for the Tri-Service VTOL Transport, Bell Model D2064. The thrust-to-weight ratio is not as good as the jet engines, but this is compensated by the much lower specific fuel consumption. For flight times in excess of about 15 minutes, the ducted fan would provide a one-man vehicle with lower gross weight at take-off. Although engines are available, the complete engine-fan-duct system would represent a new development. In addition, the magnitude of aerodynamic effects is a function of disc loading. The lower disc loading of the ducted fan would increase the aerodynamic effects, particularly the moment due to air momentum change in entering the inlet during translation velocities (discussed in detail under engine moments in the later section). Thus, it would present a more difficult development problem than the jet engine system.

D. LIQUID-FUELED ROCKET ENGINE

The use of the liquid-fueled rocket engine as a primary lift device can be eliminated for a variety of reasons. Most important of these is the requirement for development of a throttleable rocket engine with sufficient run duration. For example, a vehicle with an empty weight of 3000 pounds would require 16,000 pounds of rocket propellant. This is based on a specific impulse of 300 sec and a flight duration of 10 minutes during which the rocket supports 5/6 of the weight. The rocket engine, therefore, would have to be accurately controlled over a thrust range from 19,000 pounds to less than 3000 pounds. A throttleable rocket engine of this size would require a complete research and development program far too expensive and lengthy.

E. JET ENGINE

The earliest available lift type small jet engine is the General Electric J85. The CJ610-1 is an available commercial version without afterburner. The J85-5 is an available after-burner version. The CF700 is an aft fan version which will be available for delivery. A modification of the J85 to permit continuous vertical operation is available today. The thrust to weight ratio is sufficient to allow a feasible one man vehicle design using a single engine.

The Pratt & Whitney TJ3D-5 is a typical turbo fan, which would be required for a larger vehicle. However, two factors dictate against its immediate use. One is that it has not been modified at present for vertical operation. It might be used in a horizontal position with thrust deflection; however, this presents severe stabilization problems. Secondly, the thrust to weight ratio of a large engine is not as high as a smaller engine. Several American and British jet engines are under development with improved thrust to weight ratio and capable of vertical operation, but will not be available for several years.

F. SELECTION OF METHOD

Table II-1 presents a comparison of thrust, weight, and propellant consumption of lift engines discussed above. Any of these engines are small enough to allow for mounting a single engine on a gimbal with the vehicle structure, at the center of gravity.

The rocket engine has been eliminated because of prohibitive propellant consumption for a ten minute flight duration.

The ducted fan has been eliminated because it represents a development problem.

The jet engine approach was selected because of availability of a suitable engine at an early date with a minimum of development time and cost.

TABLE II-1

COMPARISON OF LIFT ENGINES

	Rocket	Jet (J85)			Turbofan	Ducted Fan
	Bell Agena #8096	CJ610-1	J85-5	CF700	TJ3D-5	Bell D2064
thrust, lbs	16,000	2,850	3,850	4,200	21,000	10,000
weight, lbs	296	389	538	623	4,490	2,237
T/W	54	7.32	7.16	6.65	4.69	4.48
S.F.C., lb/lb/hr	15	1	2.2	.7	.535	.15
1 min. propellant at full thrust	4000	47.5	141	49	187	25

TABLE II-2

RELATIVE MERITS OF BASIC LIFT METHODS

	Helicopter	Flying Tether	Ducted Fan	Jet Engine	Rocket Engine
Fidelity of Simulation	Poor	Poor	Fair	Good	Excellent
Availability of Hardware	Extensive Mod.Req'd.	Tether Control must be Developed	Must be Developed	Available	Limited
Cost	High	High	Medium	Low	High
Reliability and Safety	High	Low	High	High	Low
Response to Commands	Slow	Slow	Medium	Fast	Very Fast
Vehicle Weight	High	High	Low	Low	Very High
Speed and Altitude Perf.	Good	Poor	Good	Good	Excellent

Table II-2 summarizes a comparison of all basic lift methods investigated, with relative merits of each configuration in regard to several important factors.

III. CONFIGURATION

A. EVOLUTION OF CONFIGURATION

The vehicle configuration which has evolved during the course of this study is shown in Figure III-1. An artist's drawing of this configuration is depicted in the Frontispiece of this report. The basic considerations which led to this design are discussed in detail elsewhere in this report but are summarized here to give a concise picture of the compromises and decisions which were made during the course of the study.

1. The primary mission of the vehicle is to accomplish basic research on lunar landing problems as an aid in the design of the Apollo and other manned lunar landing vehicles. Thus, this research vehicle must be available soon enough so that its flight research results may be obtained before the Apollo design is completely committed. In addition, it is desirable that the cost of this first free flight lunar landing simulator be kept to a minimum. It is anticipated that eventually a larger, more complex and more expensive simulator will be required, which the first small, low cost vehicle will help to define. These requirements of early availability and minimum cost have ruled out consideration of high speed aerodynamically clean vehicles, helicopter, or ducted fan configurations which are very efficient in providing lift but which represent a long lead time and high cost development program. It was decided rather to design in the direction of a simple airframe using conventional materials, standard sizes, and conventional manufacturing techniques. The simple truss work design which has evolved provides performance capabilities adequate for the initial research tasks to be accomplished.

2. Basic to the design of a lunar simulator is a lift engine which counteracts five-sixths of earth gravity. The remaining one-sixth earth gravity will provide downward acceleration equal to lunar gravity acceleration. A turbojet engine was selected to provide this lift because of its small size, completed development status, and early availability. While for lunar simulation the jet engine need provide a lift equal only to five-sixths of the vehicle gross weight, the jet engine should have a maximum continuous thrust slightly in excess of the total gross weight, so that it can be used independently for vehicle take-off and for emergency landing. In addition, the jet engine should compensate for vehicle aerodynamic drag by tilting of the thrust vector. A single engine configuration is preferred over a dual engine configuration in order to increase vehicle reliability, and eliminate the problem of matching performance ratings of pairs of engines. Vertical mounting of the engine is preferred in order to

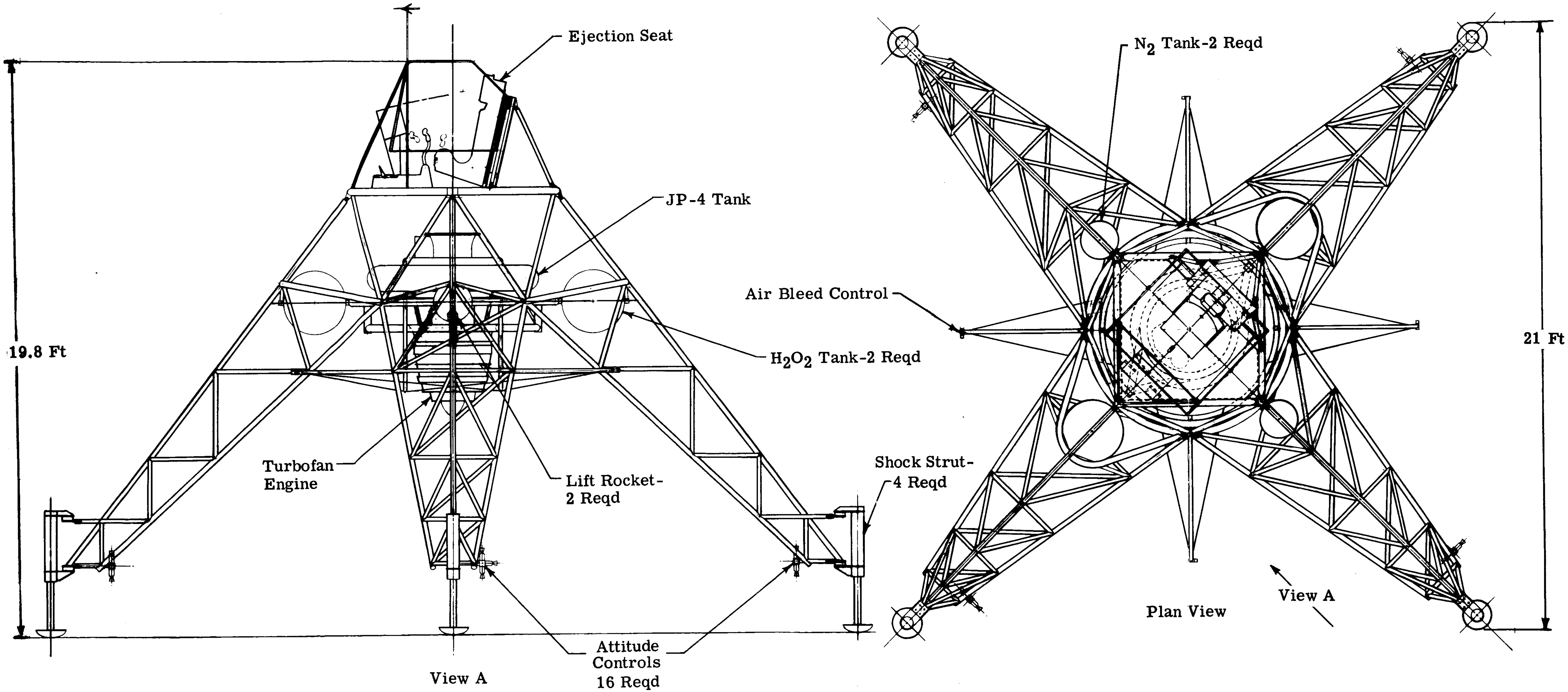


Figure III-1. Configuration Layout

eliminate thrust loss and development problems inherrent in thrust diverters. First design attempts used the General Electric J-85 engine because it has the shortest delivery time, six months. Thrust of this engine under standard sea level static conditions is 2850 lbs. It was found difficult to design a useful single engine vehicle for operation at the altitude of Edwards on a warm day, with a reasonable payload and flight duration. An attempt was made to use a J-85 engine with afterburner. However, the increased weight of the afterburner, and its high specific fuel consumption resulted in a net useful payload increase of only about 100 lbs, which was not considered to be warranted for the additional complexity involved in the operation of an afterburning engine.

The next step therefore, was to use the General Electric CF-700-2B. This is basically a J-85 engine to which has been added an air coupled aft fan. It provides 4200 lbs sea level static thrust. Delivery on this engine is eight months. Other British and American engine suppliers were contacted. It was determined that no other lift engine in this thrust range was available for delivery in less than two or three years.

It was ascertained that the eight month delivery time on the CF-700-2B is compatible with a one year delivery on the vehicle. The selection of this engine sets a ceiling on the gross take-off weight of the vehicle slightly less than the thrust available from this engine at the altitude of Edwards on a 79°F day (3840 lbs).

3. In order for the jet engine thrust vector to counteract five-sixths of earth gravity, the thrust vector must pass through the vehicle center of gravity and remain vertical, regardless of vehicle attitude. This requires that the jet engine be mounted on a two-axis gimbal and that the vehicle center of gravity be located at the intersection of the gimbal axes. This will provide neutral static stability. Shifting of equipment in the vehicle will provide a margin of either positive or negative static stability. The length of the landing legs was chosen to cause the vehicle center of gravity to fall at the intersection of the gimbal axes. This leg length also provides adequate clearance from the jet engine exhaust to the ground to prevent ground erosion. In order to prevent excessive shift in center of gravity as propellants are consumed, all propellant tanks are located in the plane of the gimbal axis. The vehicle has been designed to provide for 40° angular freedom in both axes between the jet engine and the vehicle. This allows 10° tilt of the engine to counteract vehicle drag and 30° tilt of the vehicle to accomplish translation.

4. Actual lunar vehicles will be supported by lift rockets. Thus for accurate simulation the thrust/mass/gravitational force ratios should be the same for the simulator vehicle on earth as would occur on the moon. It is possible to provide this additional lift by means of the jet engines. Pilot throttle and stick inputs could be processed through a computer and used to adjust the jet engine throttle and attitude to simulate control of lift rockets. However, because of the slower response of a jet engine, this is considered to be a poorer simulation than can be obtained by use of rocket lift engines. Since man-rated rocket systems are available today, rocket lift was selected.

A hydrogen peroxide system was selected for both lift and vehicle attitude control for which Bell Aerosystems Company has man-rated components presently available from the Mercury program, X-15, and Bell Small Rocket Lift Device. In addition, peroxide servicing equipment and experienced personnel are available at Edwards Air Force Base. A bipropellant system will provide almost twice the operating time for the same system gross weight and is presently under development. However, it was felt that this vehicle should not be a test bed for the development of a new bipropellant system. Should such a system be developed for another program, it would represent a future growth potential for this vehicle.

5. Because most of the research tasks to be accomplished at an early date require only one man, the basic vehicle has been designed as a one-man vehicle. However, for training and observation purposes, and because some tasks require a second man, the layout has been arranged to accommodate a dual seating arrangement. The weight of the second seat and man is compensated by offloading of propellants with consequently reduced flight time. The minimum performance desired with one man is ten minutes of operation of the jet engine and vehicle attitude control and two minutes operation of the lift rocket system. This will allow the vehicle to take off and climb to an altitude of 2000 ft on the jet engine, translate a distance of three miles from the take-off site, simulate two lunar landings of one minute duration each, and fly back to the take-off point. The peroxide and jet engine fuel tanks are designed to accommodate additional propellant to increase flight time approximately 50%, in the event that it is desired to use the payload allowance for additional flight duration.

6. Comparisons were made of three legged and four legged configurations. For a fixed vehicle tipover angle, the three-legged configuration requires that each leg be 41% longer than for a four-legged configuration. Because of the additional length required, and the higher bending moment resulting, each leg of the three-legged

configuration will be almost twice the weight of the four-legged configuration. Thus, the total leg weight with three legs is greater than the total leg weight for four legs. In addition, the four-legged configuration provides a convenient support for the attitude control rockets. In order to accommodate unevenness of landing terrain with four legs, each leg has been provided with long stroke shock absorbers. For transportation of this vehicle by truck or cargo aircraft, the legs are attached to the platform frame by pin joints so that the legs are easily removed.

7. The primary flight controls for vehicle attitude and jet engine throttle include a dual link, one electrical, and one direct mechanical, between the pilot controls and the attitude rockets and jet engine throttle. Although an all-electrical system would be simpler, cheaper, and lighter in weight, it was felt that this vehicle should not pioneer in the "fly-by-wire" technique, even though lunar vehicles will undoubtedly be all-electrically controlled.

8. Jet engine vertical stabilization is provided by utilization of jet engine compressor bleed air in four downward facing nozzles. Airflow from these nozzles is controlled by gyros mounted on the engine. The gimbal system is provided with a caging and locking mechanism which will effectively lock the jet engine rigid to the vehicle airframe. Thus no bleed air is required during takeoff so that full engine thrust is available. In addition, the vehicle stabilization system and the jet stabilization system can provide mutual backup to each other for greater vehicle safety in the event of failure of one of the systems.

9. All systems on the vehicle required for controllability and maneuverability are redundant with the exception of the jet engine. Methods of saving the vehicle in the event of engine failure have been examined and are discussed elsewhere in this report. However, none have been incorporated in the basic vehicle design because of the weight penalty involved.

To provide for pilot safety, a zero altitude, zero velocity, rocket type ejection seat has been provided.

10. Although this vehicle has not been designed to be aerodynamically clean, compromises have been made to reduce drag and drag moments as much as possible consistent with low cost and early availability. Tube sizes are the smallest diameter permissible for maximum slenderness ratio. The design presented is capable of full aerodynamic drag compensation for horizontal velocities up to about 60 ft/sec in any direction.

B. DESCRIPTION

The vehicle layout is shown in Figure III-1. An artist's drawing is shown in the Frontispiece. It consists of an aluminum alloy tubular truss work and platform structure, 21 ft long, 21 ft wide, 19.8 ft high, and has a gross weight of 3420 lbs.

Each leg is a tapered triangular truss of welded aluminum alloy tubing. Pin joint type attachment fittings permit removal of the legs from the platform structure for shipping. This feature will also facilitate future variations in leg geometry if desired. A landing shock absorber is attached to each leg by means of two rubber shear mounts which absorb lateral energy during landing. The shock absorbers are fitted with castered wheels. As pilot proficiency develops these will be replaced by dish or saucer type feet to afford more realistic landing simulation.

The platform structure is a welded aluminum alloy tubing circular ring at the bottom trussed to a square frame at the top. Fittings integral with the lower ring are provided for attachment of the two lower main members of each leg. Fittings are provided at each corner of the upper frame for attachment of the upper main member of each leg. Two gimbal axle housings are welded to the lower ring to permit attachment of the gimbal ring. The main lift rocket mounts are fixed to the lower ring immediately below the gimbal axle housings.

The gimbal ring is a formed steel tube containing four sets of gimbal bearings in hubs. Two sets of gimbal bearings, to which the platform structure is attached, form the outer gimbal. The inner gimbal is formed by the remaining two sets of gimbal bearings to which the engine mount is attached.

A General Electric CF-700-2B axial-flow, aft fan, jet propulsion engine modified for vertical operation supplies the main lift force for the simulator. It is supported on the gimballed engine mount (see Figure III-2),which is a welded unit fabricated of carbon steel tubing, utilizing three restraint points on the engine. One point is on the fan front frame and is capable of restraint in all three axes. Another point is on the fan front frame and is capable of vertical and lateral restraint in one axis. The third mounting point, which is on the engine front frame, acts as a stabilizer and is also capable of lateral restraint in one axis. This type of mounting system allows for thermal expansion of the engine.

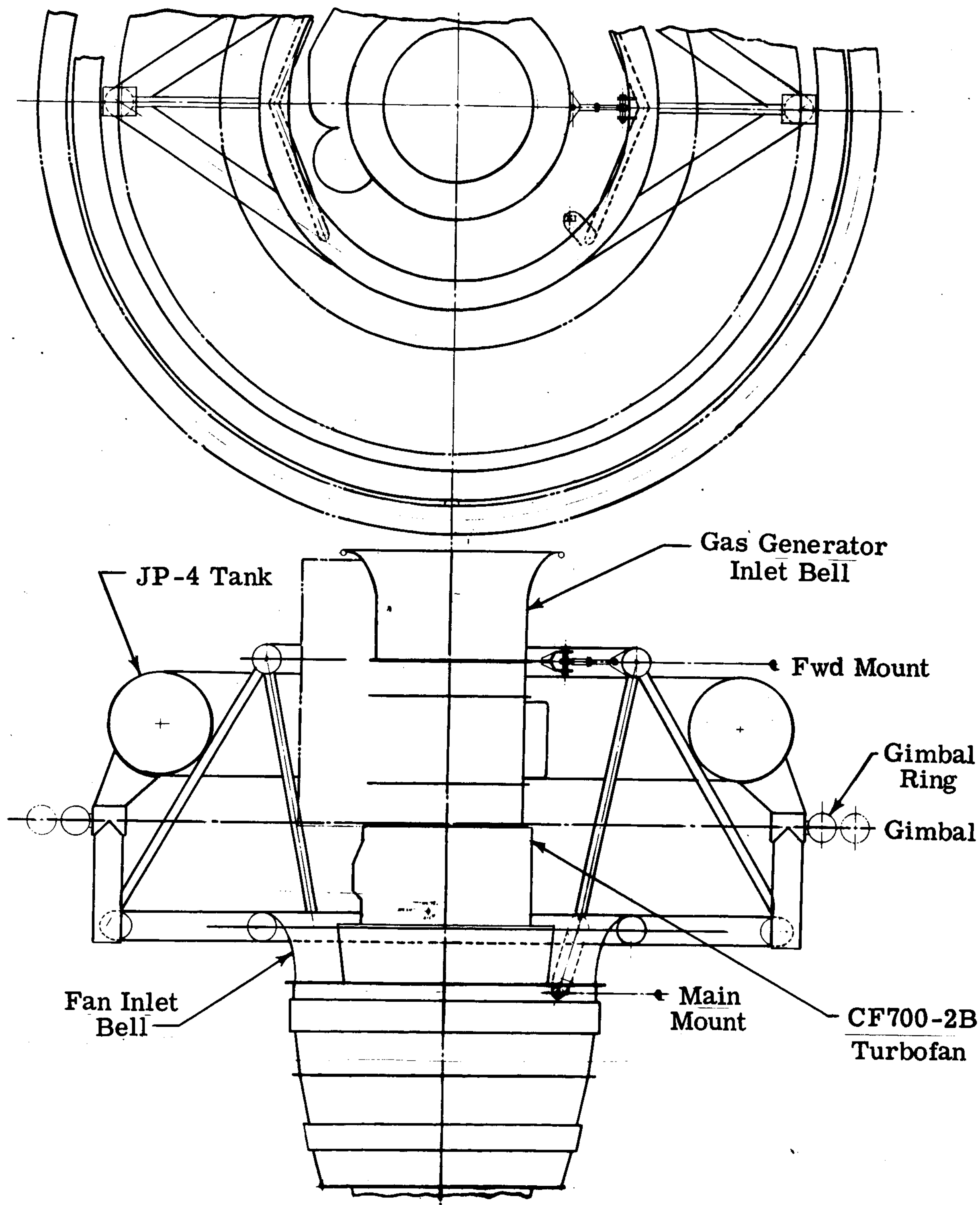


Figure III-2. Engine Mount

A toroidal tank which has a capacity of 500 lbs of JP-4 fuel surrounds the engine in the area of the compressor case and is supported by pads on the engine mount. The tank is compartmented radially into four equal segments to minimize center of gravity shift. Two fuel outlets are provided near each end of each compartment to insure uninterrupted fuel flow during maneuvering. Each pair of outlets is manifolded to a four element flow divider to maintain equal fuel withdrawal from each of the four fuel tank compartments.

Components of the jet stabilization system are secured to the engine mount structure. The jet stabilization thrust system ducts, nozzles, and valves which connect to the compressor air bleed ports on the engine are supported by structure attached to the mating flanges of the fan front and rear frames.

The gimbal pattern permits 40° tilt of the vehicle vertical axis with respect to the turbofan engine vertical axis in any direction. A gimbal centering and locking mechanism (see Figures III-3 and III-4) is provided to cage the gimballed engine for take-off or in event of malfunction of the engine stabilization system. It consists of two pairs of nitrogen actuated cylinders, one pair mounted at one of the inner gimbal bearings and the other pair mounted adjacent to one of the outer gimbal bearings. The cylinder piston rods are free floating until pressure is applied through a solenoid operated tri-port valve.

Removable floor boards of lightweight honeycomb sandwich construction and a seat support structure are fastened to the platform structure. Three sets of fittings are provided to permit either single seat or dual seat installation. Zero-zero pilot ejection seats are provided in each configuration. Aircraft type center stick and rudder pedal controls are installed as a unit to facilitate relocation for dual seat configuration.

A helicopter type instrument panel is permanently installed on the center line of the platform. Instruments included are:

airspeed indicator
altimeter
rate of climb indicator
JP-4 fuel level indicator
engine rotor speed indicator
exhaust gas temperature indicator
lube oil pressure indicator
lube oil temperature
JP-4 tank pressure

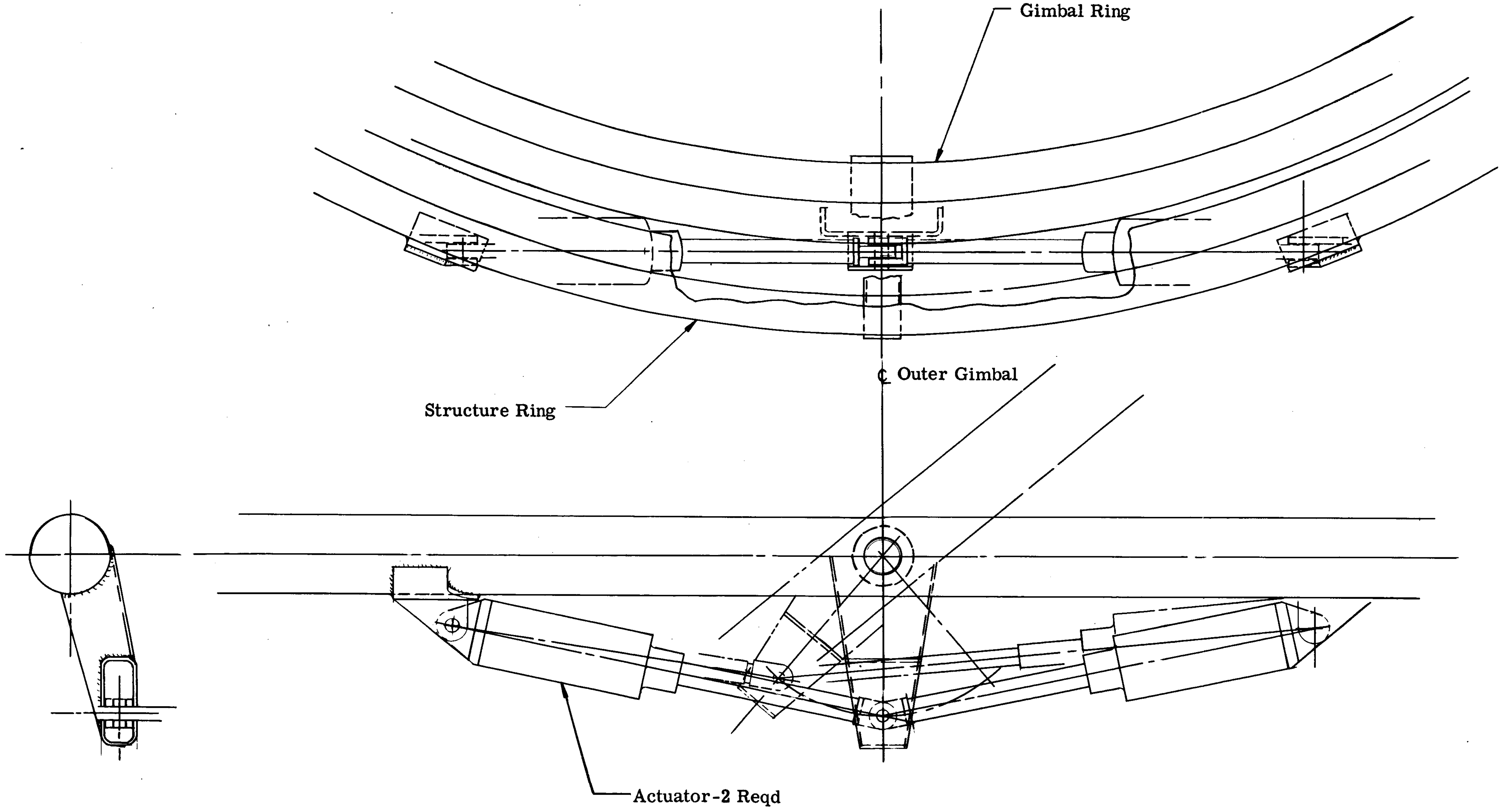


Figure III-3. Gimbal Lock

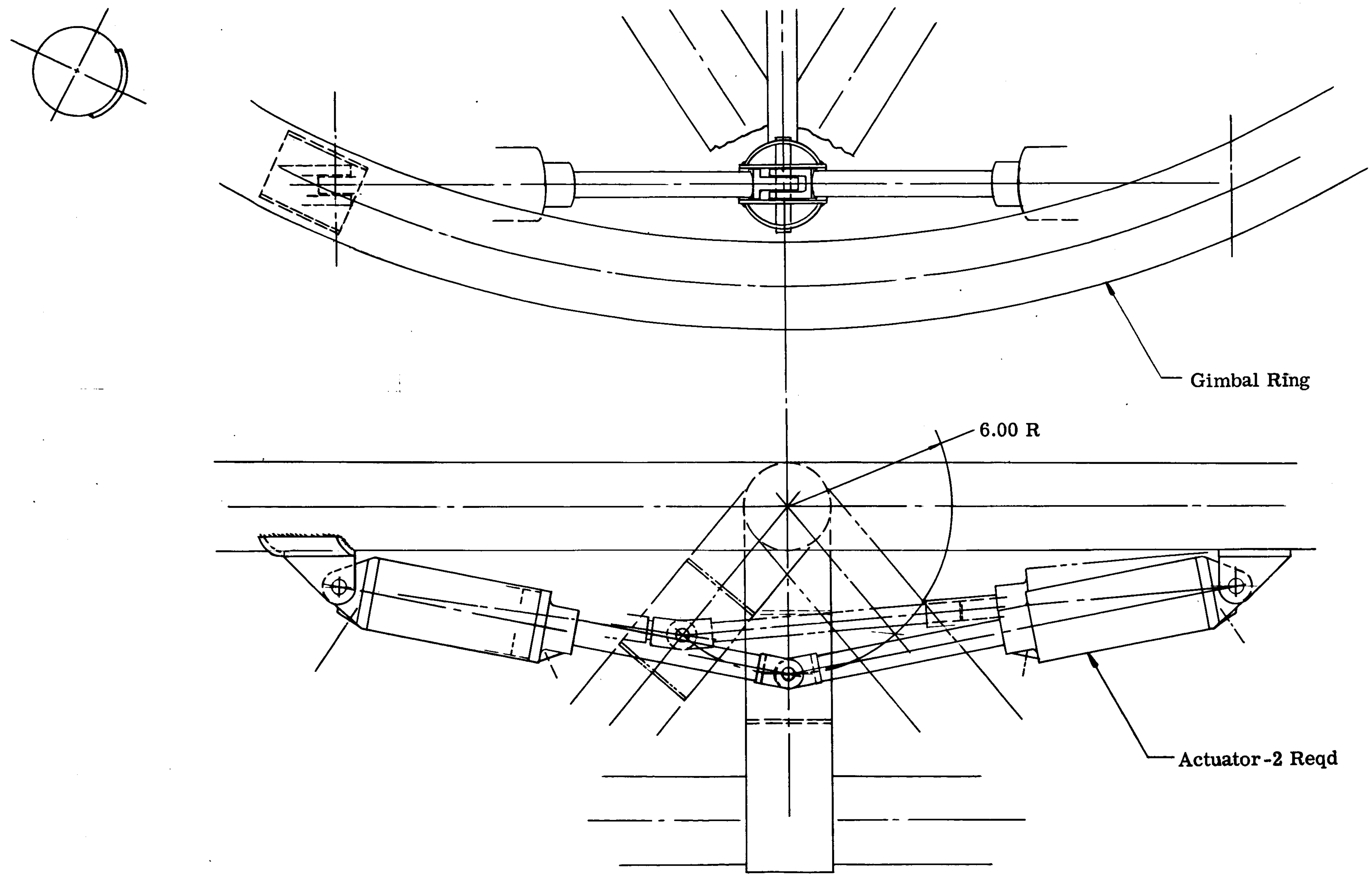


Figure III-4. Gimbal Lock

drift indicator
attitude indicator
clock
D.C. voltmeter
A.C. voltmeter
warning lights
peroxide remaining
engine thrust

The pilot enclosure consists of a windshield and side panels of tinted mylar or thin plexiglas. A pitot tube and transducer are mounted on a mast extending above one corner of the windshield. The side panels are hinged to permit ingress to the seat. Pilot oxygen for total flight duration is supplied from a 24 cu ft (270 cu in. @ 2250 psi) air bottle.

Two 500 lb throttleable thrust lift rockets are mounted fixed vertically to the underside of the platform lower ring diametrically opposite each other. Four clusters of four 80 lb (max) thrust attitude control rockets are mounted on the lower end of each leg inboard of the landing shock strut. Two 400 lb capacity H_2O_2 propellant tanks are trunnion mounted within the upper trusswork of two opposite legs centered on the vertical center of gravity of the vehicle. Within the trusswork of the other two legs are two N_2 spheres, also trunnion mounted, for pressurization of the rocket propulsion systems.

Components will be installed on the vehicle wherever possible to attain the desired balance. Figure III-5 is a block diagram of the interconnections between the installed systems. Each system is discussed in detail in other sections of this report.

C. DESIGN CRITERIA AND LOADS

1. General

The following summarizes the results of a brief study of structural design criteria for a Lunar Landing Simulator (BAC Model 7161).

Selection of Limit Sink Speed:

This problem has been examined from two points of view:

a. what sink speed could we consider to be adequate for a lunar landing vehicle?

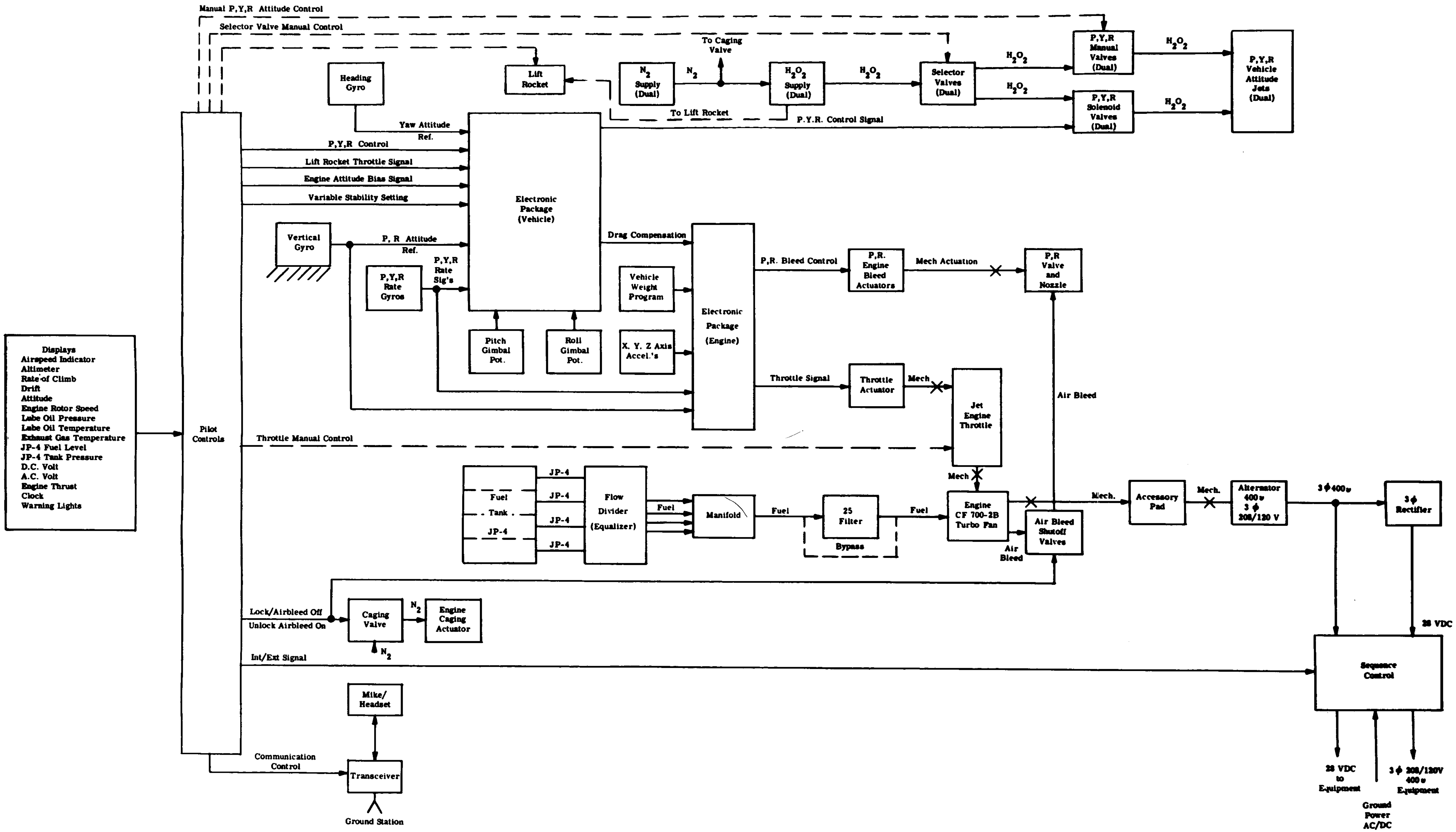


Figure III-5. System Interconnection Block Diagram

b. what sink speed could we consider to be adequate for the proposed Lunar Landing Simulator in the event of a jet engine flameout "near" the ground, i.e., so near to the ground that safety devices could not be actuated before contact with the ground was made?

With regard to (a), a "feel for the problem" has been obtained by performing the following analysis. The lunar vehicle has been assumed to hover above the ground just prior to setting down on its landing gear. This would correspond to the operator's looking over the immediate terrain on which the vehicle would come to rest. From this height a descent is initiated by retarding rocket engine thrust at a linear rate until ground contact is made. This procedure is believed realistic as far as initiating the descent is concerned, but it is undoubtedly very conservative with regard to the operator's continuing to retard thrust until ground contact is made. This assumption is made, nevertheless, since it simplifies the resulting analysis and, also, since it approaches the problem from a conservative direction.

Figure III-6 summarizes the results of the analysis. The ordinate defines the height above the ground at which thrust retardation is initiated. The abscissa defines, in effect, the rate of thrust retardation. Time to complete throttle retardation has been specified for this purpose since it describes the situation more clearly than per cent thrust reduction per second.

A "throttle completely retarded at impact" boundary is shown since the present analysis is not valid to the left of this line. Lines of constant touchdown velocity (V_I) are shown in the figure.

Examination of the curves reveals that touchdown speeds of 10 to 12 ft/sec provide for fairly rough handling of the lunar vehicle by the operator. For example, starting at a height of 20 ft, the operator can retard thrust at a rate which leads to zero thrust at the end of six seconds without exceeding a touchdown speed of 10 ft/sec. Since the operator is not apt to continue to reduce thrust at this rate for the full six seconds, the touchdown speed would normally be less than 10 ft/sec.

With regard to (b) above, it was assumed that flameout of the simulator's jet engine occurred at a linear rate for a period of one second and that the fixed rocket engine thrust equalled 0.16 times the weight of the simulator. It was found that the contact velocity would be 13.5 ft/sec from a drop height of 4.5 ft. From this rough

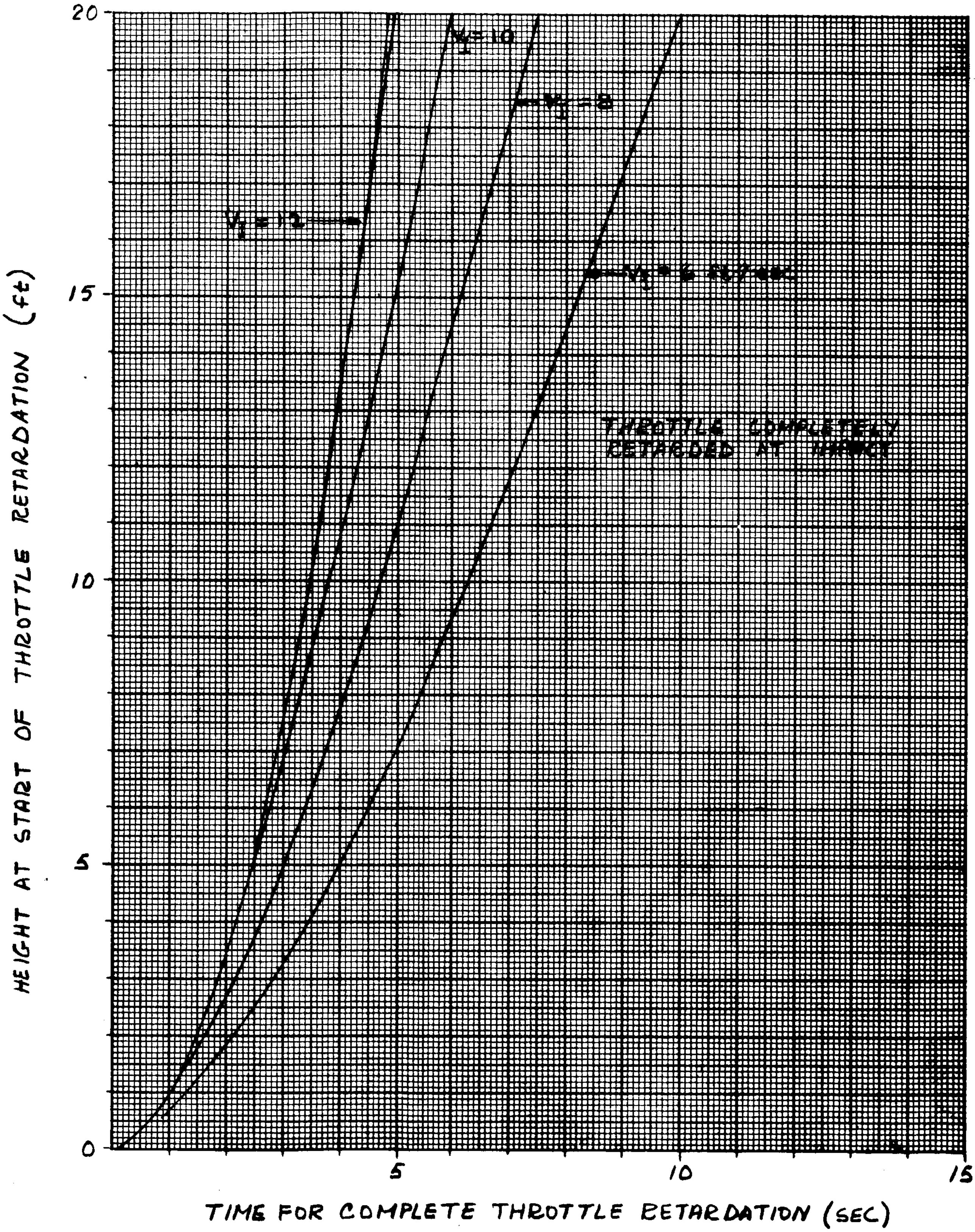


Figure III-6. Lunar Landing Impact Velocities, V_I, for Initial Descent Heights and Throttle Retardation Rates analysis, it is evident that an engine flameout from even modest heights results in very high ground contact velocities. Design of the landing gear shock absorbing system with sufficient margin of strength to cope with engine flameouts from substantial heights (10 ft or more) is undoubtedly impractical. A relatively high sink speed is nevertheless desirable and a 10 ft/sec limit sink speed is recommended for design of the simulator. For purposes of simulation, a lower sink speed could realistically be considered. The above value is recommended, however, in order to help assure safe operation of the simulator during training operations when high rates of thrust retardation may be applied.

Selection of Other Landing Parameters:

The percentage of lift acting during landing (two-thirds of the simulator weight) has been selected on the basis of the helicopter landing requirements of MIL-S-8698(ASG).

The sink speed assumed to apply during landings on rough terrain or under side drift (windy day) conditions is 60% of limit sink speed, or six ft/sec. This reduced value of sink speed is predicated upon the assumption that landings upon rough terrain or on windy days will be undertaken only after familiarization flights have been completed under more ideal landing conditions.

The side drift velocity of 3 ft/sec which is proposed for design is based upon the assumption that the operator of the simulator will compensate for side drift due to whatever average wind prevails at the time of landing. Under these conditions, drift of the simulator will be due primarily to fluctuations in the average wind which will be substantially less than the average wind itself.

The design weight for landing is taken as the maximum takeoff weight, i.e., design gross weight, since it is anticipated that flight durations may be very short under some operating conditions, particularly during training periods.

2. Design Criteria

The following loading conditions are recommended for design of the Lunar Landing Simulator.

The landing weight shall correspond to the design gross weight of 3400 lbs.

a. Limit Loads

Loads derived from the following loading conditions are limit loads unless otherwise specified. When subjected to limit loads the simulator shall not experience deformations which will interfere with its operation, or require replacement or repair of its structure due to permanent distortions.

b. Ultimate Loads

Ultimate loads are limit loads multiplied by the ultimate factor of safety of 1.5. The structure of the simulator shall not fail to sustain ultimate loads.

c. Landing Conditions

(1) Level Landing, Smooth Terrain

All landing legs shall contact the ground simultaneously at the limit sink speed of 10 ft/sec. Total vertical thrust shall equal two-thirds of the design gross weight.

(2) Level Landing, Irregular Terrain

The landing legs shall contact the ground at a sink speed of six ft/sec in whichever attitude requires maximum energy absorption from a single landing gear shock absorber. The total vertical thrust shall equal two-thirds of the design gross weight. The height differential between the highest and lowest ground contact locations shall be three feet.

(3) Side Drift Landing

The simulator shall be trimmed so as to produce zero side drift in a 10 mph wind and shall contact the ground with a side drift velocity of ± 3.0 ft/sec. The sink speed shall be six ft/sec. The ground contact attitude shall be such that:

(a) one leg contacts first with the lateral ground reaction applied in the plane of the leg and the simulator center of gravity.

(b) alternatively, two adjacent legs contact first with the lateral ground reaction at each leg being equal and perpendicular to a line drawn between the ground contact points.

(4) Crash Landing

The operator's seat, safety belt, shoulder harness, and all attachments for components adjacent to the operator shall not fail when subjected to an ultimate inertia loading of 40-g applied within a 20 degree cone aligned to the vertical axis. Items of equipment and elements of the propulsion and control systems shall be attached in such a manner that they do not present a hazard to the pilot during a 40-g impact.

d. Miscellaneous Conditions

(1) Propulsion and Control Systems

All propulsion and control systems components and their attachments shall withstand the inertia forces associated with the loading conditions of c(1), c(2) and c(3) above, in any normal operating position.

(2) Operator's Controls

The operator's controls shall conform to the loading requirements of paragraph 3.7 of MIL-A-8865(ASG).

3. Loads

A preliminary study has been made of the landing shock absorbers and the results are summarized in Table III-1. These can be considered as minimum requirements for the design of the shock absorber and preliminary load conditions for the design of the airframe. In all conditions, conservative (low) values of efficiency were assumed so that any changes resulting from detail design of the gear would tend to reduce loads or increase allowable sinking speeds. The calculations are based on a shock absorber design having an available stroke of 17 inches and a compression ratio of 6.25. The factors considered in the selection of the shock absorber characteristics are explained in Section III.E.

TABLE III-1

DESIGN VERTICAL GROUND REACTIONS

Legs	Condition	Load Factor			V	h	Efficiency	Stroke
		Engine	Legs	Total	Ft/Sec	Ft		Inch
2	Limit	.67	1.33	2.00	6	1.67	.60	14.3
4	Limit	.67	2.66	3.33	10	4.65	.70	12.3
2	Ult.	.67	2.00	2.67	11.1	5.72	.80	15.1
4	Ult.	.67	4.00	4.67	12.5	7.35	.65	13.0
2	Limit	0	1.33	1.33	0	-1.13	.75	14.3
4	Limit	0	2.66	2.66	5.9	.55	.55	14.3
2	Limit	.167	1.33	1.50	7.3	5.00	.65	14.3
4	Limit	.167	2.66	2.83	11.0	11.25	.75	12.3

D. STRUCTURAL DESCRIPTION AND ANALYSIS

The primary structure consists of a platform, four legs, and an engine mount and gimbal ring assembly. The platform is a cage-like structure surrounding the jet engine, and the four legs radiate outwards from the platform periphery as shown in Figure III-1. All the equipment is mounted on the platform or the legs except the jet engine, the jet fuel, and the jet engine stabilization system which are supported by the engine mount. The engine mount pivots on the gimbal ring and the gimbal ring pivots on the platform. These two pivot axis are at right angles and in the neutral position they are both horizontal, thus permitting the jet engine to be tilted in any direction relative to the platform up to an angle of 40^{o} from the vertical. In actual flight the jet engine is maintained in a vertical attitude and the platform is tilted to simulate the motion of a lunar landing vehicle which tilts to translate. Lift rockets are fixed relative to the platform and thus provide a horizontal component of thrust when the platform is tilted. Monopropellant reaction control rockets at the ends of the legs provide the attitude control forces for the platform, and bleed air jets attached to the engine provide the attitude control forces for the engine. At the discretion of the pilot the jet engine can be rigidly aligned to the platform in the neutral position by a gimbal locking system, and the entire vehicle controlled by the reaction control rockets on the legs. The cockpit is mounted above the platform in a fixed position relative to the platform.

The structure of the platform is a tubular truss cage of 6061-T6 aluminum alloy. Main members are a ring at the bottom approximately 86 inches in diameter, and a square at the top approximately 61 inches on a side. These are connected by vertical and diagonal members running from the corners of the upper square to strategic spots around the circumference of the lower circle thus forming a cage. The top of the cage is closed by the cockpit floor in the plane of the upper square, but the bottom of the cage is left open to permit the jet engine to project through. The gimbal pivot axis is in the plane of the lower circle.

Tube size for the sides of the square at the top of the platform is 3-1/2 x .083. These members are designed to support cockpit inertia loads in bending combined with axial compression produced by the ultimate 4 leg level landing condition. Tube size for the lower ring is 4 x .083. This is critical for the radial loads from the legs produced by the ultimate 4 leg level landing condition. The vertical and diagonal tubes between the upper and lower cage members are 2 x .049. This size is chosen to blend well with the upper and lower cage members at the welded joints. Therefore, all of the vertical and diagonal members of the cage are overstrength.

A study of the typical leg shown in Figure III-7 has been completed and the results are summarized in Table III-2. Two designs are shown: one with the minimum weight tubes for each member and one with the smallest practical diameter tubes for each member. The latter design results in lower aerodynamic drag which is an important consideration for the legs, since the center of pressure of the basic vehicle is below the gimbal axis. The weight shown in Table III-2 is indicative of the vehicle leg weight and shows that the weight estimate of Section III.F is conservative.

An IBM 704 digital computer program now in existence at Bell Aerosystems Company could be used for loads in the leg and cage truss members for a final detail design. This program, utilizing the method of direct stiffness calculations, develops five major items of information in the process of solution. These are:

1. Displacement influence coefficients

2. Displacements due to given applied load or temperature conditions

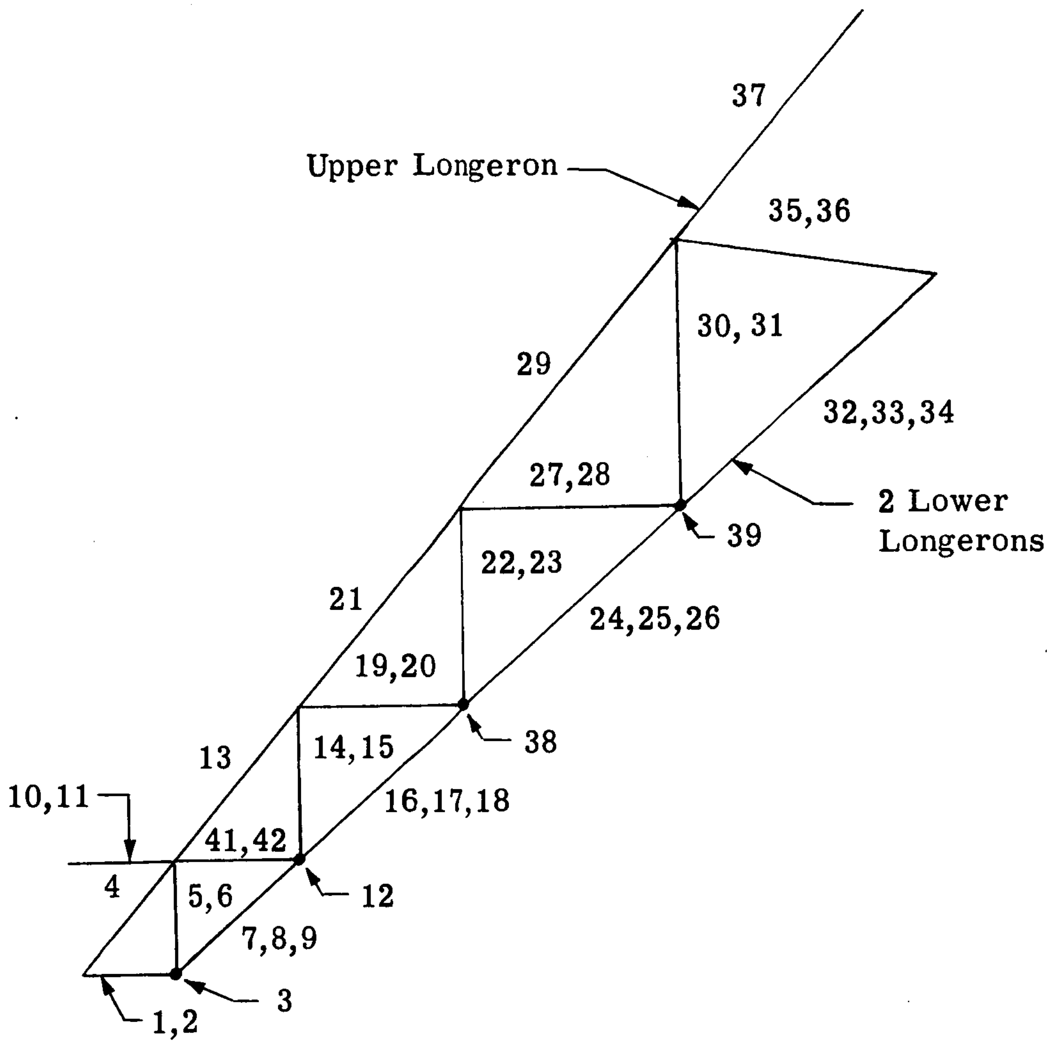
3. Internal stress influence coefficients

4. Internal stresses due to given applied load or temperature conditions

5. External support reactions

By use of this program, it is possible to solve quickly various truss configurations and thus optimize design.

The engine mount is a welded tubular steel truss. The configuration shown in Figure III-2 is the result of a study of several schemes and is a simple, compact arrangement producing a minimum of interference with the fan inlet air. Steel is used primarily for the ease of attaching steel fittings at the pivot points and engine mounting points. Stiffness is essential in the engine mount so no weight saving is possible with aluminum. This is also true of the gimbal ring which is a 3-inch diameter, .083 wall steel tube, although the same stiffness could be achieved using a larger diameter aluminum tube. A simple study is required to determine the optimum size and material of the gimbal ring but the weight used in Section III.F is representative and conservative.



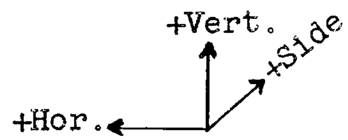
7,18,26,34 — Diagonals — Lower Truss
3,12,38,39 — Cross Members — Lower Truss
Odd Numbers Shown, Even Numbers are Behind Odd.

Figure III-7. Support Leg — Truss Nomenclature

TABLE III-2

ULTIMATE MEMBER LOADS, TUBE SIZES AND WEIGHTS PER SUPPORT LEG

Tube Loads { + Tension / - Compression }



Cond. I - +3200 lbs vert.
Cond. III - { +2055 lbs vert. / -1200 lbs hor. / ±1200 lbs side }
Applied at ground reaction point

Member	Cond. I Vert. lb	Cond. III Vert. lb	Cond. III Hor. lb	Cond. III Side lb	Cond. III Comb. lb	L in.	Min.Wt.Tube Size for Crit. Cond. in.	Wt lb	Design Tube Size for Crit. Cond. in.	Wt. lb	
4	- 3970	-2550	- 80	0	-2630	21.0	1-5/8 x .028	.298	2-1/4 x .095	1.605	Upper Longerons
13	- 8210	-5290	+4440	0	- 850	29.0	2-1/4 x .049	.992	2-1/4 x .095	2.220	
21	-10030	-6460	+4690	0	-1770	39.6	2-1/2 x .049	1.512	2-1/4 x .095	3.025	
29	-11320	-7290	+4850	0	-2440	54.5	3 x .049	2.500	2-1/4 x .095	4.160	
37	-12180	-7830	+4960	0	-2870	53.6	3-1/4 x .049	2.662	2-1/4 x .095	4.100	
8 & 9	+ 2100	+1550	-2560	±8050	-9060 +7040	24.5	2-1/4 x .049	.838	1-3/4 x .095	1.221	Lower Longerons
16 & 17	+ 3480	+2240	-2710	±4830	-5300 +4360	33.4	2 x .035	.729	1-3/4 x .095	1.665	
24 & 25	+ 4220	+2710	-2810	±5260	-5360 +5160	45.9	2-1/4 x .035	1.135	1-3/4 x .095	2.288	
32 & 33	+ 4770	+3060	-2880	±5500	-5320 +5680	52.7	2-1/4 x .049	1.802	1-3/4 x .095	2.627	
7	0	0	0	±2660	-2660 +2660	28.7	1-1/2 x .028	.376	1-3/8 x .058	.695	Diagonals Lower Truss
18	0	0	0	±2620	-2620 +2620	39.2	1-3/4 x .035	.747	1-3/8 x .058	.949	
26	0	0	0	±1956	-1956 +1956	53.6	1-3/4 x .035	1.021	1-3/8 x .058	1.297	
34	0	0	0	± 864	- 864 + 864	64.5	1-5/8 x .028	.916	1-3/8 x .058	1.561	

Total (Page 1) 20.032 lbs (Page 1) 35.214 lbs
Total (Page 2) 8.914 lbs (Page 2) 11.072 lbs
Total = 28.946 lbs Total = 46.286 lbs

TABLE III-2 (continued)

Member	Cond. I Vert. lb	Cond. III Vert. lb	Cond. III Hor. lb	Cond. III Side lb	Cond. III Comb. lb	L in.	Min. Wt. Tube size for Crit. Cond. in.	Wt lb	Design Tube size for Crit. Cond. in.	Wt. lb	
3	- 227	- 146	- 256	±1528	-1930 +1126	13.0	1 x .028	.112	1-1/8 x .035	.157	
12	- 125	- 80	+ 304	±4830	-4606 +5054	17.4	1-3/4 x .035	.332	1-1/8 x .058	.343	
38	- 22	- 14	+ 10	±1190	-1194 +1186	23.8	1 x .028	.206	1-1/8 x .035	.288	Cross Members Lower Truss
39	+ 83	+ 53	- 8	± 855	- 810 + 900	32.4	1 x .028	.280	1-1/8 x .035	.392	
1 & 2	+1837	+1180	-1955	±5450	-6225 +4675	14.1	2 x .035	.308	1-1/8 x .065	.308	
5 & 6	+1776	+1140	+1925	±6450	-3385 +9515	18.1	1-1/8 x .083	.497	1-1/8 x .083	.497	
10 & 11	- 552	- 355	+1295	±5310	-4370 +6250	31.2	2 x .035	.680	1-1/8 x .095	.967	
41 & 42	+1470	+ 945	-1538	±4260	-4853 +3667	19.7	1-7/8 x .035	.402	1-1/8 x .058	.388	
14 & 15	- 785	- 504	+ 109	±1810	-2205 +1415	24.6	1-1/4 x .028	.266	1-1/8 x .035	.298	Diagonals Side Trusses
19 & 20	+ 606	+ 390	- 80	±1190	- 880 +1500	26.9	1 x .028	.232	1-1/8 x .035	.326	
22 & 23	- 526	- 338	+ 100	±1326	-1564 +1088	33.6	1-1/4 x .028	.354	1-1/8 x .035	.406	
27 & 28	+ 444	+ 286	- 58	± 864	- 636 +1092	37.0	1 x .028	.320	1-1/8 x .035	.447	
30 & 31	- 409	- 263	+ 52	± 972	-1183 + 761	46.1	1-3/8 x .028	.551	1-1/8 x .049	.774	
35 & 36	+ 297	+ 191	- 38	± 548	- 395 + 701	44.2	1 x .028	.382	1-1/8 x .035	.535	
							Total = 8.914		Total = 11.072		

E. LANDING GEAR CONFIGURATION

The landing gear consists of four conventional vertical shock struts, one on the end of each leg. The design characteristics of the shock absorber are given in Table III-1. Each strut is attached to the leg by rubber shear pads in an arrangement which permits horizontal deflection of the landing foot at the lower end of the shock strut. The shear pads are designed to allow a 6.75 inch lateral deflection of the landing foot for a side drift velocity of 3 feet per second with one leg receiving the full impact. This produces a lateral load factor of .50. When two legs impact simultaneously the deflection is 4.75 inches and the load factor is .71. These values are chosen so that the relative magnitude of the loads in the leg truss members is the same for the side drift landing mode as for the design vertical landing conditions shown in Table III-2.

The design of the vertical shock struts is dictated by the desire to produce a relatively small ground reaction factor while still accommodating the static ground conditions. The strut must not be fully compressed when two diagonally opposite legs are supporting a fully loaded vehicle nor fully extended when all four legs are supporting an empty vehicle. With such a wide variation in static ground conditions, it is necessary to use a high compression ratio. This results in high ground reaction factors if the entire available stroke is utilized for the design landing condition. The solution is to provide a sufficiently long stroke so that only a portion of the available stroke is utilized for the design landing conditions, thereby maintaining low compression ratios and corresponding load factors during landing. This relatively inefficient use of the shock absorber results in an efficient overall vehicle design since the structural loads are kept to a minimum and the weight is concentrated in the landing gear shock struts. If a smaller shock strut were used with correspondingly higher load factors, the vehicle structure would be heavier, and ballast would be needed in the vicinity of the shock struts to keep the center of gravity at the gimbal plane, thereby making the overall vehicle heavier.

The static ground reaction characteristics of the strut were chosen as follows: strut compressed to 1.3 times fully extended pressure when the empty vehicle rests on four legs; and strut compressed to 90% of available stroke when the fully loaded vehicle rests on two diagonally opposite legs. With an empty weight of 2100 lbs and a fully loaded weight of 3400 lbs, this gives a compression ratio of 1.3 for empty vehicle on four legs; 2.1 for full vehicle on two legs; and 6.25 for a fully compressed strut.

The required length of total stroke is based on the design condition of the fully-loaded vehicle landing on two diagonally opposite legs with a six teet per second vertical sinking speed. Assuming an efficiency of only .60 for this condition, it is possible to keep the ground reaction factor down to 1.33 by utilizing only 14.25 inches of stroke of a strut with a total available stroke of 17 inches. The characteristics of this shock absorber were calculated for other landing conditions using conservative estimates of efficiency and the results are tabulated in Table III-1.

Alternate gear arrangements considered were (1) a vertical shock strut rigidly mounted to the leg, (2) a jointed leg as shown in Figure III-9, and (3) a leg pivoted at the attachment of the two lower longerons to the platform, using a shock absorber in place of the upper longeron in the top bay. With the latter two of these alternate arrangements, some combinations of vertical and side reactions would cause the resulting line of action to pass below the pivot point and above the vehicle center of gravity. No compression of the shock absorber would occur and the leg would be overloaded. (If the line of action passes below the center of gravity the vehicle will turn over, no matter what the shock absorber configuration.) The first alternate configuration will not absorb the horizontal component of kinetic energy in a side-drift landing, which would likewise lead to failure of the leg. Another undesirable feature of the second or third alternate configurations is that the landing foot must scrub outward in order to move upward during shock absorber action. In a symmetrical landing on a soft or rough surface which would tend to prevent the scrubbing motion the induced horizontal loads on all legs would balance against each other and the motion of the shock absorbers would be thus restrained, thereby imposing higher than permissible vertical load factors on the vehicle.

F. WEIGHT AND BALANCE SUMMARY

Table III-3 is a summary of the weight and balance for the free flight lunar landing simulator. It indicates the following weights:

gross at take-off	3440 lbs
landing	2540 lbs
empty	2013 lbs

Table III-4 is a summary of the balance of the vehicle exclusive of the gimbal mounted engine assembly. The shift in center of gravity of this part of the vehicle as propellants are consumed is a measure of degree to which neutral static stability can be maintained. It will be noted that the total vertical travel is .9 inches or $\pm$.45 inches around a middle point. Analog simulation studies indicate that the pilot is unaware of shifts of up to two inches.

TABLE III-3

WEIGHT AND BALANCE SUMMARY

	Weight	Horizontal		Lateral		Vertical	
		Arm*	Moment	Arm*	Moment	Arm*	Moment
Structure	(632)						
Truss - Legs	250	200.0	50000	200.0	50000	90.5	22625
Gimbal and Bearings	213	200.0	42600	200.0	42600	136.4	29053
Truss - Main	100	200.0	20000	200.0	20000	158.0	15800
Platform, Support & Floor	34	194.0	6596	200.0	6800	185.1	6293
Misc. Supports, etc.	35	196.9	6892	200.0	7000	172.3	6031
Landing Gear	150	200.0	30000	200.0	30000	28.6	4290
Propulsion	(1022)						
Engine	623	198.2	123479	199.4	124226	129.0	80367
Alternator	32	188.1	6019	205.8	6586	143.0	4576
Engine Accessories	43	200.0	8600	200.0	8600	124.2	5341
H_2O_2 System	226	198.9	44951	200.0	45200	113.7	25561
H_2O_2 System Mounts	25	200.0	5000	200.0	5000	130.0	3450
Engine Fuel Tank	50	200.0	10000	200.0	10000	148.0	7400
Engine Plumbing	16	200.0	3200	200.0	3200	148.0	2368
Engine Controls	7	200.0	1400	200.0	1400	148.0	1036
Electrical	22	200.0	4400	200.0	4400	180.0	3960
Flight Controls	30	174.4	5232	200.0	6000	192.4	5772
Instruments	(30)						
Engine	10	175.0	1750	200.0	2000	212.0	2120
Flight	20	175.9	3518	196.4	3928	215.2	4304
Furnishings	(107)						
Seat	92	211.2	19430	200.0	18400	204.9	18851
Oxygen bottle	15	200.0	3000	200.0	3000	190.0	2850
Communication	20	230.0	4600	207.0	4140	189.0	3780
Weight Empty	2013		400667		402480		255828
Crew	200	207.5	41500	200.0	40000	206.0	41200
Payload	200	200.0	40000	200.0	40000	155.0	31000
Propellant	(1000)						
Jet Engine Fuel	400	200.0	80000	200.0	80000	148.0	59200
Rocket Propellant	600	200.0	120000	200.0	120000	134.5	80700
Pressurization Gas	27	200.0	5400	200.0	5400	138.0	3726
Useful Load	1427		286900		285400		215826
Gross Weight	3440	199.9	687567	200.0	687880	137.1	471654
Propellant							
Jet Engine Fuel	-360	200.0	-72000	200.0	-72000	149.0	-53640
Rocket Fuel	-540	200.0	-108000	200.0	-108000	135.2	-73008
Landing Weight	2540	199.8	507567	200.0	507880	135.8	345006

*The horizontal reference plane is a plane 200 inches forward of the plane passing thru the of the side legs of the vehicle.
The lateral reference plane is aplane 200 inches to the left of the plane passing thru the of the forward and aft legs of the vehicle.
The vertical reference plane is the plane 138 inches below the center line of the gimbal.

TABLE III-4

WEIGHT AND BALANCE SUMMARY

The calculation below is the weight and balance of vehicle less the gimbal ring and the associated parts attached to it. These items are the engine and its accessories, the jet fuel tank, the jet fuel, and the necessary mounting for these items.

	Weight	Horizontal		Lateral		Vertical	
		Arm	Moment	Arm	Moment	Arm	Moment
Gross Weight	3440	199.9	687567	200.0	687880	137.1	471654
Less:							
Engine Gimbal Mount	-213		-42600		-42600		-29053
Engine	-623		-123479		-124226		-80367
Alternator	- 32		- 6019		- 6586		- 4576
Engine Accessories							
Inlet Duct	- 9		- 1800		- 1800		- 1446
Inlet Bullet	- 1		- 200		- 200		- 156
Fan inlet Duct	- 17		- 3400		- 3400		- 2137
Air Bleed (4)	- 16		- 3200		- 3200		- 1600
Engine Fuel Tank	- 50		-10000		-10000		- 7400
Jet Engine Fuel	-400		-80000		-80000		-59200
Weight (not including gimbal and associated parts)	2079	200.5	416869	200.0	415868	137.4	285719
Less:							
Rocket Propellant	-250		-50000		-50000		-34535
Flight Weight (Vertical C.G. at lowest water line)	1829	200.6	366869	200.0	365868	137.3	251184
Less:							
Rocket Propellant	-290		-58000		-58000		-38509
Landing Weight (including 10% residual)	1539	200.7	308869	200.0	307868	138.2	212675
Gimbal Ring Station						138.0	

Moments of inertia of the Lunar Landing Simulator are as follows:

	Takeoff	Landing
I_x =	2474 slug ft^2	2473 slug ft^2
I_y =	2827 slug ft^2	2551 slug ft^2
I_z =	3165 slug ft^2	2662 slug ft^2

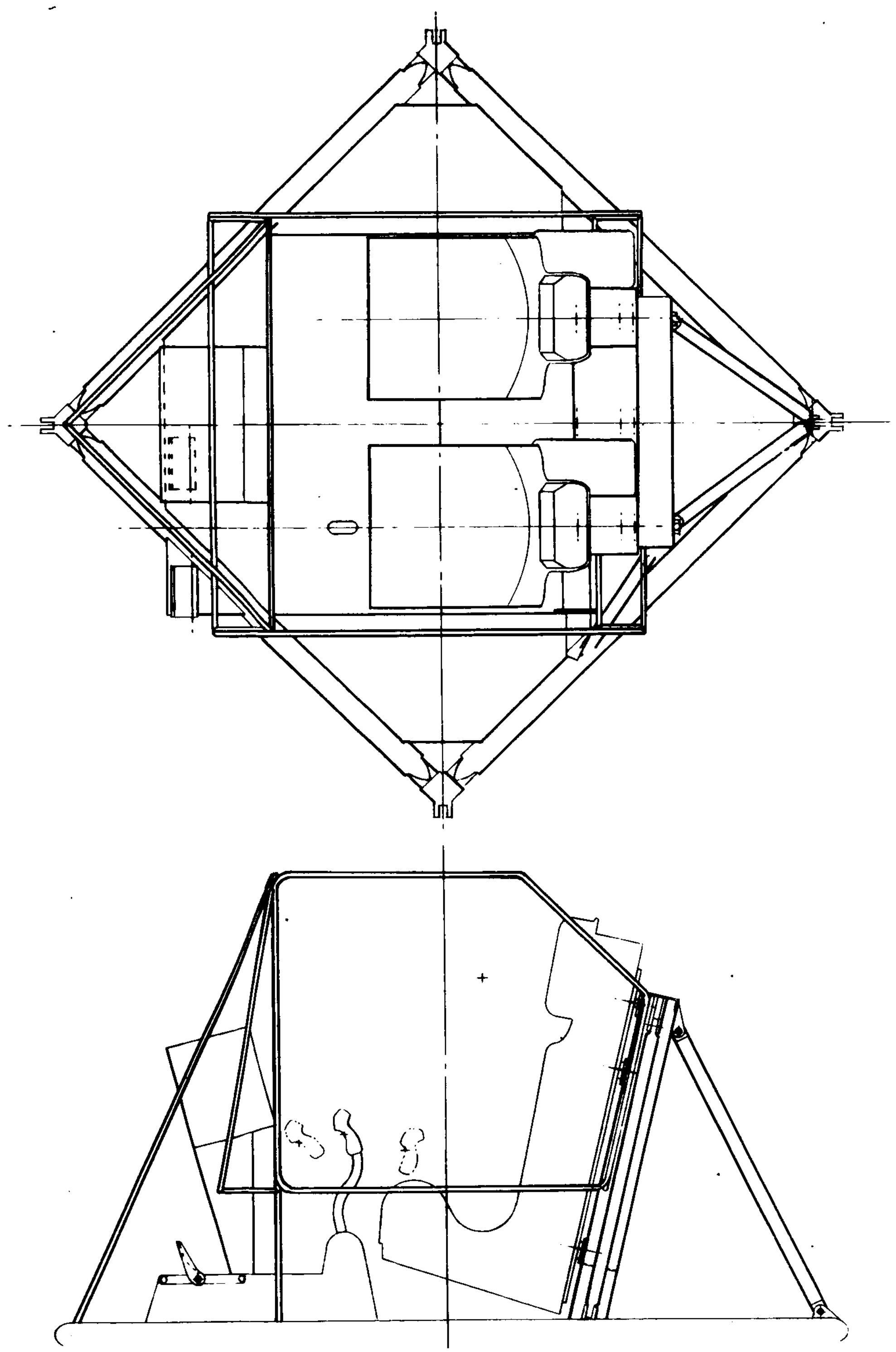
I_x and I_y do not include the gimbal ring and the associated parts attached to it. These items are the engine and its accessories, the jet fuel tank, the jet fuel, and the necessary mounting of these items.

G. DUAL AND HORIZONTAL SEATING

Dual seating (see Figure III-8) may be obtained by minor re-arrangement of the simulator platform equipment. To convert the single seat configuration to accommodate two men, the existing seat support can be utilized. Two sections of floor panels adjacent to each side of the ejection seat are removed prior to seat relocation. The pilot's seat, control stick, and rudder pedal assembly is relocated approximately 12.0 inches to the left of the vehicle center line. A second ejection seat is mounted to the seat support approximately 12.0 inches to the right of the vehicle center line.

To accommodate research on manual flight control with the pilot seated in various attitudes from a vertical to a horizontal position, an adjustable reclining seat can also be provided in either a single or dual seat arrangement. Since flying in a horizontal position may be more difficult, for flight safety reasons the reclining pilot should be backed up by a co-pilot in an erect seat. Thus, the reclining seat shown in Figure III-8 is a dual seating arrangement with one seat erect. The reclining seat is trunnion mounted close to its center of gravity. A "fly by wire" side arm controller is mounted to the arm rests so as to move with the seat. The reclining seat is fitted with a manually fired cartridge to rotate the seat to an erect position in the event that emergency ejection is required.

For dual seating, propellants are offloaded approximately 30% as follows:


Seats Erect

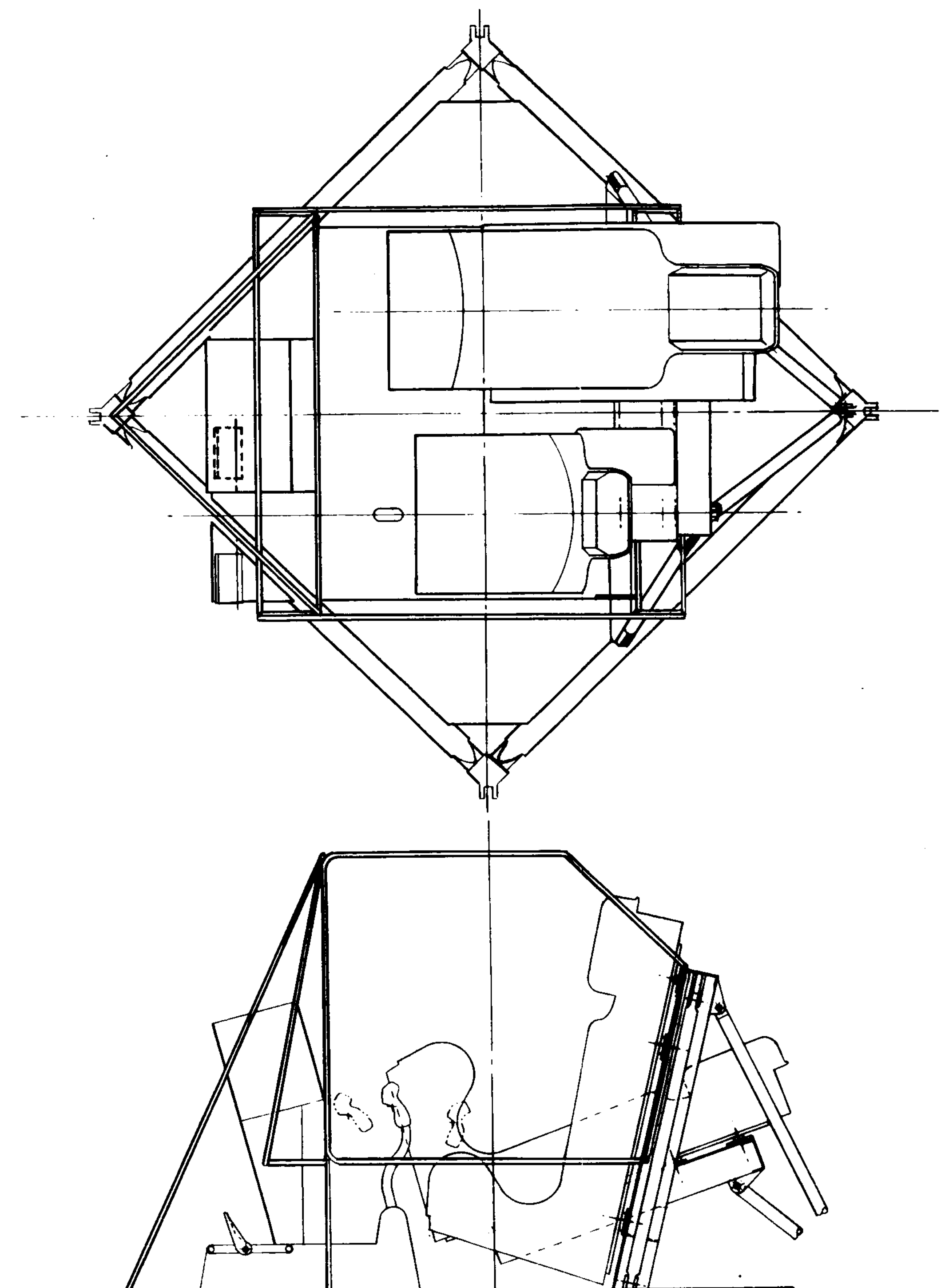

One Seat Erect & One Reclining

Figure III-8. Dual Seating Arrangement

Gross Weight One-Man Version			3440
Plus:		+307	
2nd Man	+200		
Seat	+ 92		
O_2 Bottle	+ 15		
Minus:		-307	
Jet Fuel	-102		
Rocket Propellant	-205		
Gross Weight Two-Man			3440

In order to preserve vehicle balance at the intersection of the gimbal axes, it will be necessary to relocate other payload at a position lower down the leg structure, when dual seating is used.

H. ALTERNATE CONFIGURATIONS

During the evolution of the design presented in this report, other configurations were examined and rejected. Drawings of these configurations and reason for the rejection of each is presented here. These drawings are not complete since each concept was carried only far enough to ascertain feasibility or desirability of the concept.

1. CJ-610 (J-85) Engine (Figure III-9)

Initial design studies of the simulator were based on the use of a single General Electric CJ-610 turbojet engine. This engine has a takeoff thrust rating of 2850 lbs at standard sea level static conditions. Although the desired balance was achieved with this configuration, the weight of the vehicle was estimated at 2707 lbs. This is well beyond the lift capability of the CJ-610 engine on a warm day at an altitude of 2000 ft. Although the addition of an afterburner to this engine would increase the takeoff thrust about 1000 lbs, the additional hardware weight, the high specific fuel consumption, and the large shift in engine center of gravity all resulted in a net increase in payload capability of only about 100 lbs.

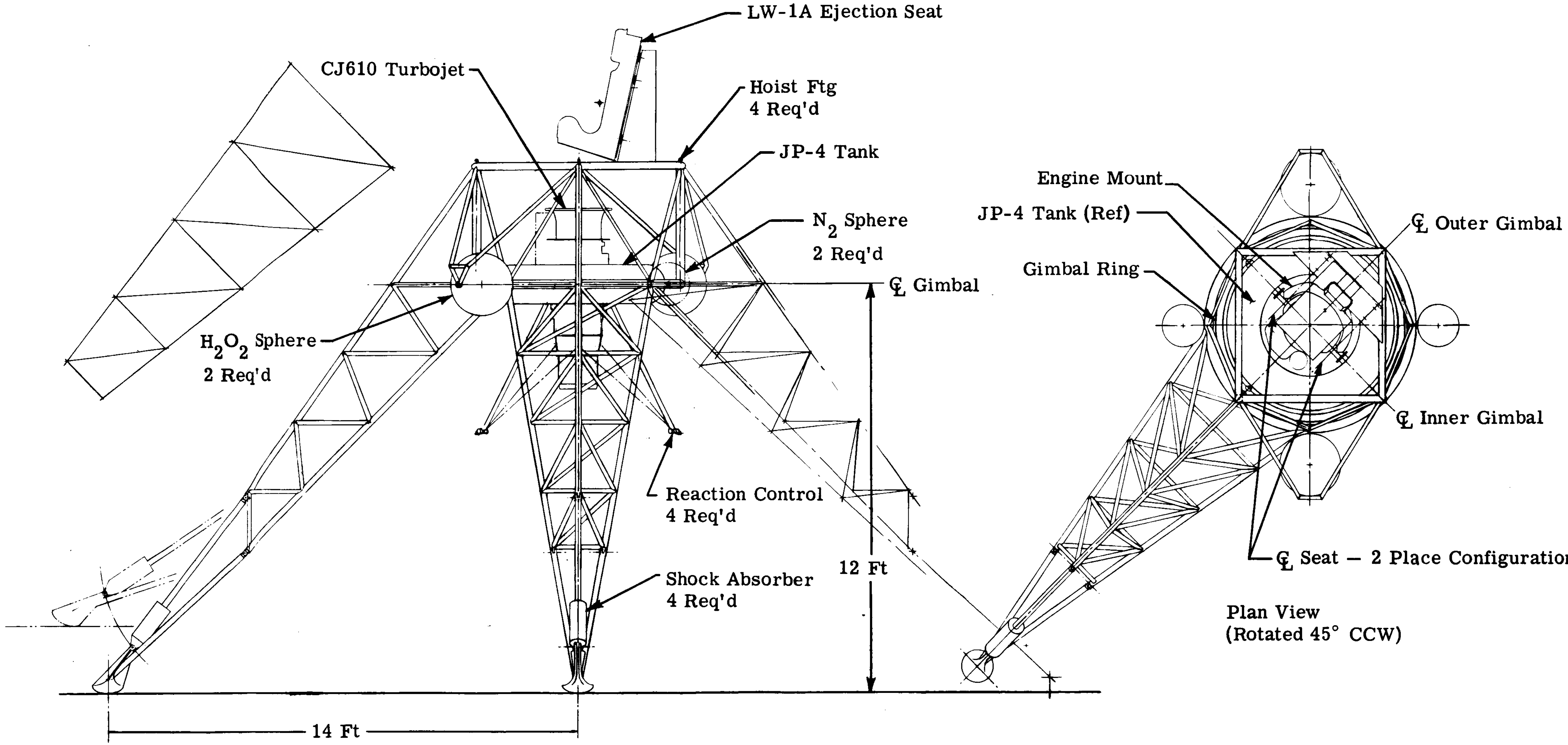


Figure III-9. Configuration Using CJ-610 Engine

2. Twin J-85 Engines (Figure III-10)

The twin engine configuration has thrust more than adequate to meet all payload and flight envelope requirements, even on a hot day. However, considering the availability of the CF-700-2B engine, which has adequate thrust for the job and a considerably lower SFC, the two J-85 engine configuration was abandoned because of its reduced reliability and increased operational problems of balancing performance of two engines.

3. Single Tube Leg

Although no layout was made, a study was conducted to compare the drag and weight of a single tube leg versus the truss design. It was determined that a 10-3/4" outside diameter, 7075-T6 aluminum alloy tubing with .080" thickness had equal strength to the truss construction. Drag is approximately equal to the truss leg but weight is calculated to be 50% heavier, therefore, this single tube design was rejected.

4. Jet Fuel Tanks Outside of Gimbal Ring (Figure III-11)

Two different approaches were tried to replace the toroidal jet fuel tank with spherical tanks. The first approach was to place spherical tanks on the vehicle structure outside of the gimbal rings. Flexible lines are required to transfer the fuel from the tanks around the gimbal bearings to the engine. Because a smaller gimbal ring could be used and because the spherical tanks are more efficient than the toroidal tank, a decrease in weight was realized by this configuration. However, the large increase in polar moment of inertial due to moving the fuel mass farther from the center of gravity eliminated this configuration from further consideration.

5. Spherical Jet Fuel Tanks Inside Gimbal Ring (Figure III-12)

A design which replaces the toroidal tank with four spherical fuel tanks clustered around the engine requires a change in the gimbal ring from a circular to a square shape. The increased structural weight is prohibitive.

6. Jointed Legs (Figure III-9)

Jointed legs with shock absorbers were compared with fixed vertical shock absorbers on rigid legs. This configuration was examined because it appeared to give greater vertical motion to the landing foot on impact, with a small shock absorber travel. However,

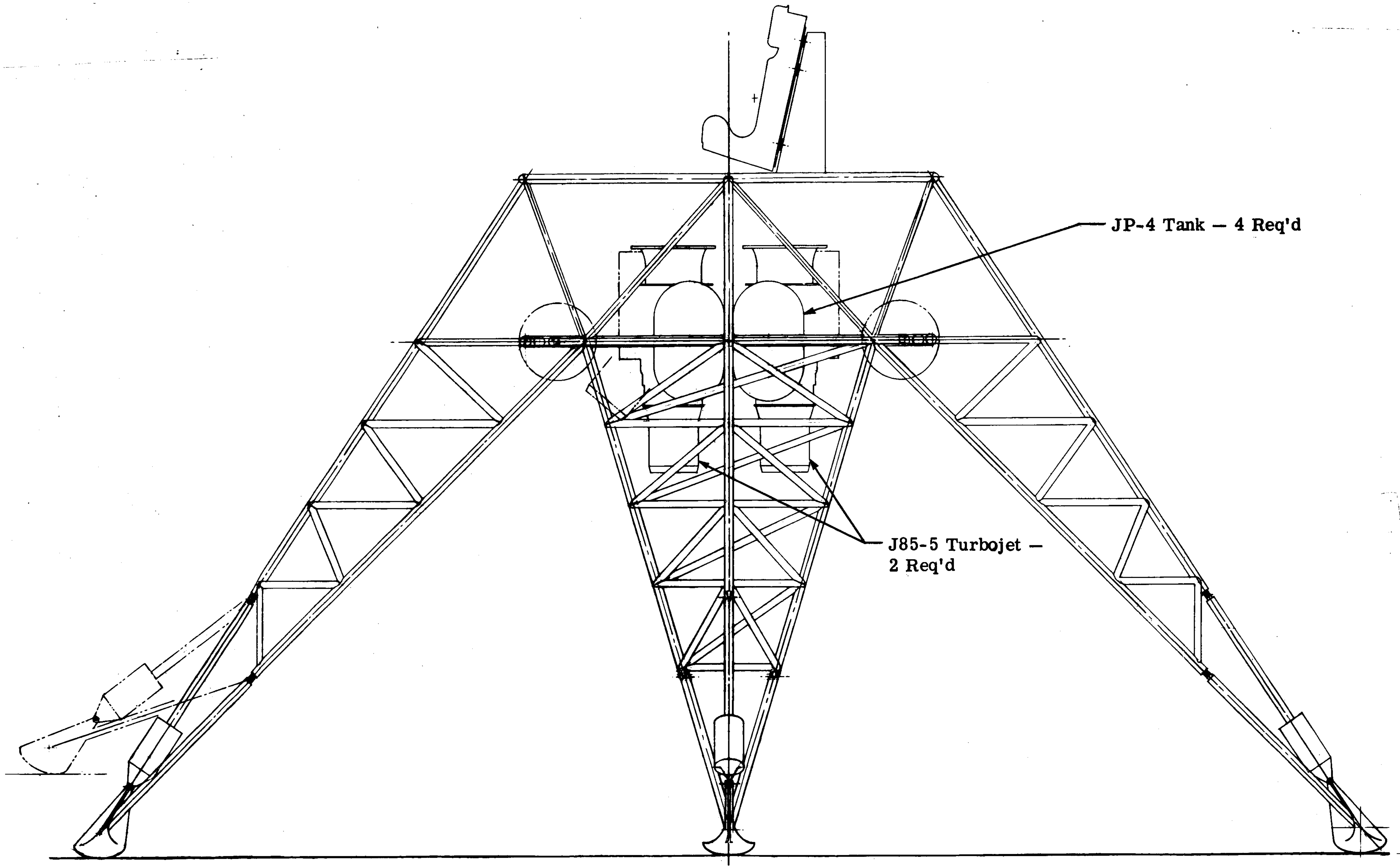


Figure III-10. Configuration Using Twin CJ-610 Engines

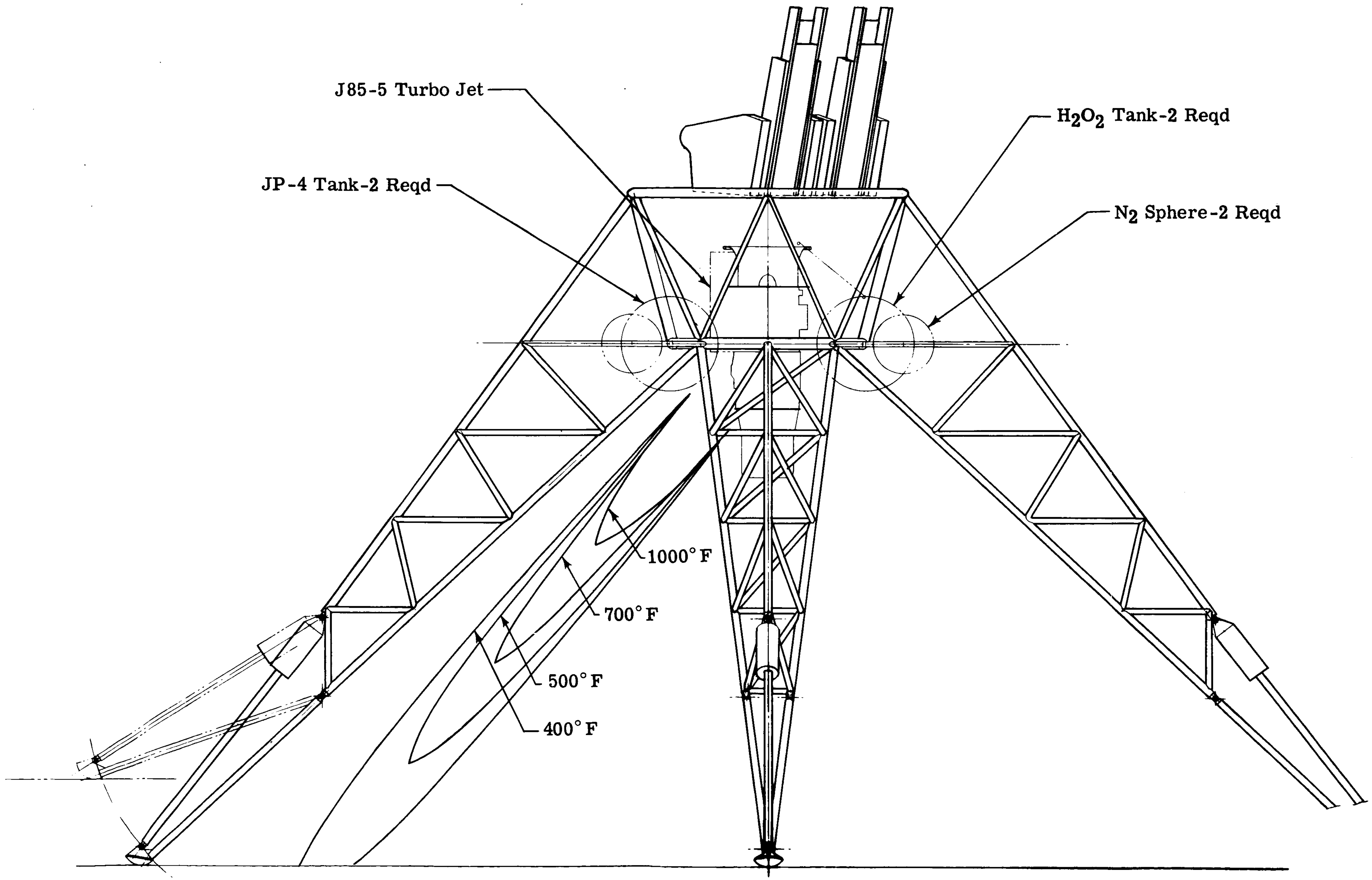


Figure III-11. Configuration with JP-4 Tanks Outside Gimbal Assembly

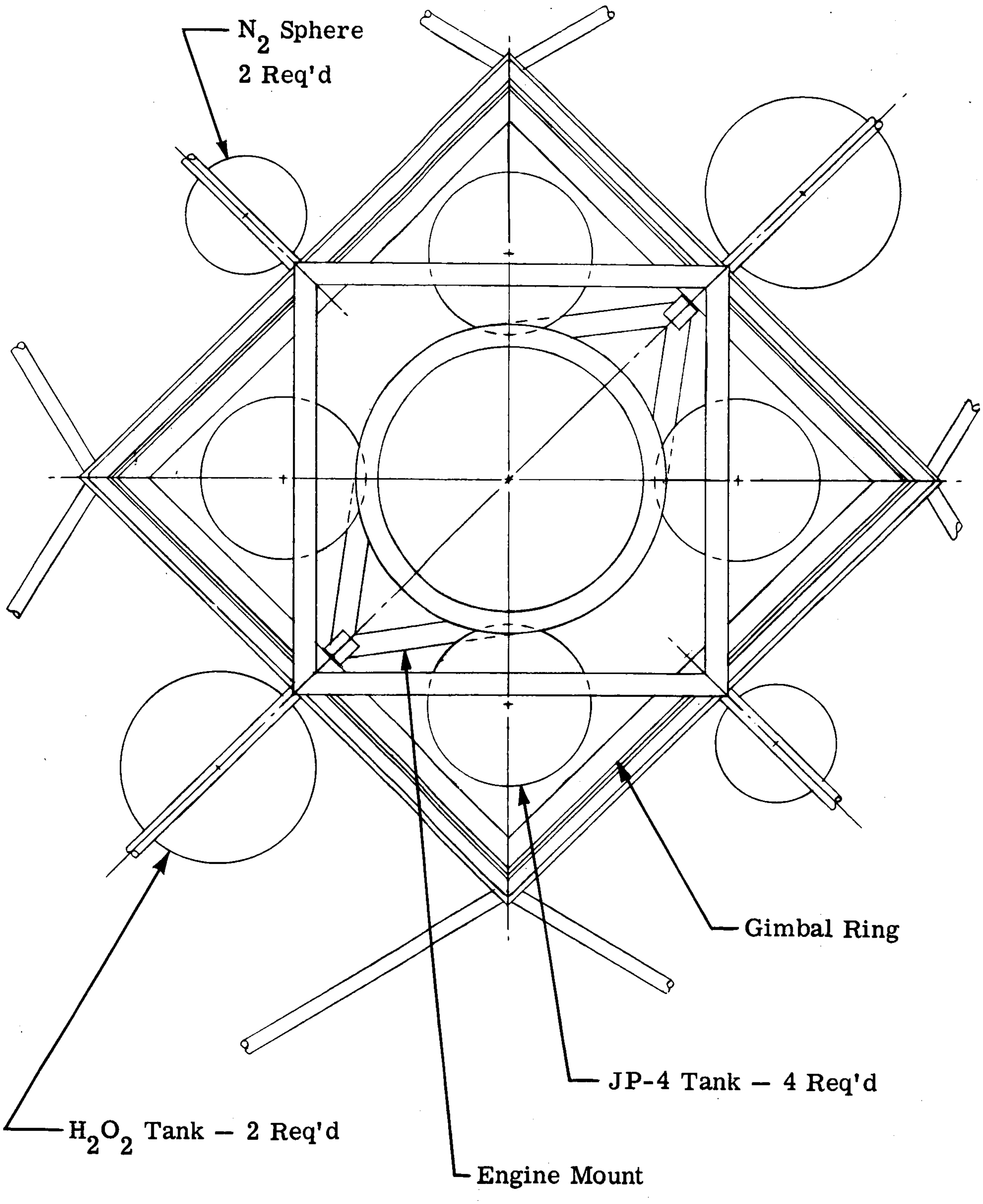


Figure III-12. Configuration Using Spherical JP-4 Tanks the analysis described in Section III.E of this report indicated that under certain impact conditions, no shock absorption action would take place and intolerable shock loads would result on the rigid structural members.

This design was replaced by the currently proposed shear mounted vertical shock struts.

I. JET ENGINE DESCRIPTION

The General Electric CF700-2B is an axial-flow, aft fan, jet propulsion engine. It incorporates an 8-stage, axial-flow compressor driven by a two-stage reaction turbine; an annular combustion section; a free-floating, single-stage aft fan; a fixed-area, concentric exhaust section and an integrated control system. Designed for commercial or military applications, the CF700F-2B offers a high thrust-to-weight ratio coupled with low specific fuel consumption.

One feature of the engine, and perhaps the most important one from an operating standpoint, is its low specific fuel consumption. The inclusion of the aft fan component, which increases mass airflow and decreases jet velocity, gives the CF700-2B a large increase in thrust over that available from a comparable turbojet, while consuming the same amount of fuel. This factor significantly increases the mission capability of modern jet aircraft.

The bypass or fan type engine, which combines the best elements of jet and propeller operation, has long been accepted as a means of increasing turbojet performance. General Electric design engineers, by perfecting the aft fan which is aerodynamically, but not mechanically connected to the engine rotor, have solved the problems associated with the use of the fan principle. Through the use of the free floating, single-stage fan, optimum fan performance is realized without compromising the performance or significantly changing the basic gas generator.

Another feature of the engine that contributes to improved aircraft mission capability is its light weight. This is made possible primarily by efficient design of structures and the use of sheet steel construction.

The high pressure-ratio, single-spool gas generator compressor incorporates variable inlet guide vanes and variable interstage bleed at low speeds, enabling the engine to make rapid stall free accelerations with a single rotor in the gas generator.

The engine uses the same gas generator as the General Electric J-85, thereby providing the same ease of inspection, assembly and disassembly and the same economy of maintenance and overhaul as the J-85. The reliability of the components has been proved by many hours of flight and factory test.

IV. STABILITY AND CONTROL

This section describes the requirements, operation and design of the control systems used to: a) tilt the vehicle; b) attitude stabilize the jet engine; c) automatically throttle the jet engine; d) automatically compensate for drag and changes in vehicle weight; and e) provide adjustable stability to the vehicle attitude control system.

Vehicle attitude is controlled by hydrogen peroxide reaction jets operated by the pilot. Lateral translation is accomplished by tilting the vehicle and hence, tilting the thrust vector of the lift rockets. Vertical translation is accomplished by pilot control of the lift rocket throttle. The vehicle is attached to the engine through a two-axis gimbal which effectively isolates vehicle pitch and roll attitude motions from jet engine motions. A rigid connection exists between the vehicle and the engine in the yaw axis, since heading changes do not normally affect the verticality of the engine. If the vehicle is tilted when a change in heading is commanded, the jet engine bleed air control system will maintain the verticality of the jet engine by applying moments about the pitch and roll gimbal axes.

Attitude stabilization of the jet engine is provided by autopilot control of reaction jets,using engine compressor bleed air. The engine thrust vector is nominally vertical except for the tilt required to compensate for the lateral drag force on the vehicle.

Jet engine thrust is controlled by programming throttle setting to maintain thrust to weight ratio of 5/6 as jet and rocket propellants are expended. Vehicle vertical drag component is compensated by measuring drag and commanding throttle change to compensate.

A. DESCRIPTION OF CONTROL MODES

Centering actuators on the two gimbals between the vehicle and engine provide a capability of flying and landing the lunar vehicle in the following modes:

1. Jet engine lift only with gimbals caged or uncaged.

2. Jet engine plus rocket lift with gimbals uncaged.

3. Emergency backup modes.

1. Jet Engine Only

Operation with the jet engine supplying all of the lift permits the conservation of rocket fuel during the portions of flight which do not require simulation of lunar conditions. Operation in this mode of flight can also be used as a backup in the event of a lift rocket failure.

With the gimbals caged, the attitude of the jet engine can be controlled with the hydrogen peroxide reaction jet attitude control system. Lateral translation is accomplished by tilting the vehicle and therefore the thrust vector of the engine. Vertical translation is accomplished by pilot control of the engine throttle. The engine bleed air nozzle attitude control system can be used instead of the hydrogen peroxide jet controls, to control vehicle attitude.

The engine bleed-air attitude control system is capable of imparting angular accelerations on the engine and vehicle up to 0.4 radians/second^2 about the pitch and roll axes. Control about the yaw axis in this mode would still be with the hydrogen peroxide reaction jets.

With the gimbals uncaged, the jet engine is stabilized to the vertical by the engine bleed air control system; and the attitude of the vehicle is controlled with the hydrogen peroxide jets. Lateral translation is accomplished by pilot control of the engine attitude with the engine bleed air controls. Vertical translation is by pilot control of the engine throttle.

2. Jet Engine Plus Rocket Lift

Operation with the jet engine supporting five-sixths the vehicle weight and the lift rockets supporting the remaining one-sixth of the weight is used when lunar gravity and vacuum conditions are simulated.

In this mode of operation the gimbals are uncaged. The throttle of the jet engine is automatically controlled about the 5/6-g nominal setting to compensate for changes in vehicle weight and for the vertical drag force on the vehicle. The engine bleed air control system automatically tilts the jet engine about the pitch and roll axes to compensate for lateral drag on the vehicle. The pilot operates the hydrogen peroxide jets to tilt the vehicle which also tilts the lift rockets to accomplish lateral translations. The cosine loss of vertical thrust is compensated by pilot control of the lift rocket throttle. Vertical translation is accomplished by pilot control of the lift rocket throttle.

3. Emergency Modes

In the event of a failure in the lift rocket system, the pilot shuts down the lift rocket system and takes over control of the jet engine throttle. Flight then continues as described under jet engine only control.

Should a failure occur in the hydrogen peroxide control system, the pilot shuts down the failed system and continues to operate with the redundant hydrogen peroxide system.

Should a failure occur in the engine bleed air control system, the pilot shuts down the bleed air control syste, cages the gimbals and continues to operate with the hydrogen peroxide control system.

In the event of a failure in the jet engine throttle servo control system, the pilot manually overrides the servo with his throttle quadrant and disconnects the throttle servo system.

A failure in the jet engine requires the pilot to initiate emergency ejection.

Should a jet engine stabilization system failure occur calling for engine attitudes greater than 15 degrees from vertical, or greater than 10 degrees from vertical in conjunction with a 5 degree per second rate, the bleed air main valve will be shut off, a warning light will notify the pilot, and the gimbal caging actuators will be energized. The pilot will then control vehicle attitude with the hydrogen peroxide control system. Redundant attitude and rate sensors are used for reliable indication of this type of failure.

Should a vehicle electrical attitude control system failure occur which calls for attitudes greater than 45 degrees from vertical, or greater than 30 degrees from vertical in conjunction with a 5 degree per second rate, a warning light will notify the pilot. The pilot will then control vehicle attitude with the manual control system. Because the pilot is in the vehicle, it is expected that his attitude and rate sensing capabilities will be adequate backup for indication of this type of failure.

The above failure modes will be checked out thoroughly during operational tests of the jet engine and vehicle stabilization systems. Preliminary tests on jet engine and vehicle controls will be made with the vehicle tethered.

B. CONTROL REQUIREMENTS

1. Vehicle Attitude Control

A major objective of the lunar landing simulator flight program is to determine the levels of control power and damping that would be necessary and desirable for actual piloted soft lunar landing missions. This objective demands that the lunar landing simulator have variable stability and control power which can cover a sufficient range to establish optimum levels. As an initial basis for establishing these control power and damping levels, the results of studies on VTOL systems - in and around hover - were examined. Results of the more pertinent VTOL studies are summarized in Figures IV-1, IV-2, and IV-3. These curves show boundaries of satisfactory and unsatisfactory control power and damping determined in these hovering and low-speed flight tests. The X-14 flight test and Faye's (Reference A) simulator results shown in the figures, define the boundaries between satisfactory and marginally acceptable control based on a Cooper rating value of 3.5. The Tapscott curves (Reference B) also shown are minimum acceptable boundaries based on pilot opinion from a variable stability helicopter flight test program. The fairly large difference in acceptable boundaries noted from each of these studies must be attributed to the following factors: 1) the maneuvering task performed in the study, 2) speed stability of the vehicle (M_u), which determines the sensitivity of the vehicle to gusts, and 3) degree of simulation realism. More recent studies (unpublished) conducted at Princeton University on a variable stability helicopter have demonstrated the great importance of vehicle speed stability, M_u, (change in pitching moment per unit change in forward speed) on the control power and damping boundaries. Their results show that increasing M_u shifts the acceptable boundaries to higher levels of control power and damping. This speed stability effect (M_u) could explain the higher control power boundaries determined by Tapscott. Since the intent of the proposed simulator is to

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Reference A - Faye, Alan E. Jr., "Attitude Control Requirements for Hovering Determined Through the Use of a Piloted Flight Simulator". NASA TN D-792, 1961.

Reference B - Salmirs, Seymour and Tapscott, Robert J., "The Effects of Various Combinations of Damping and Control Power on Helicopter Handling Qualities During Both Instrument and Visual Flight". NASA TN D-58, 1959.

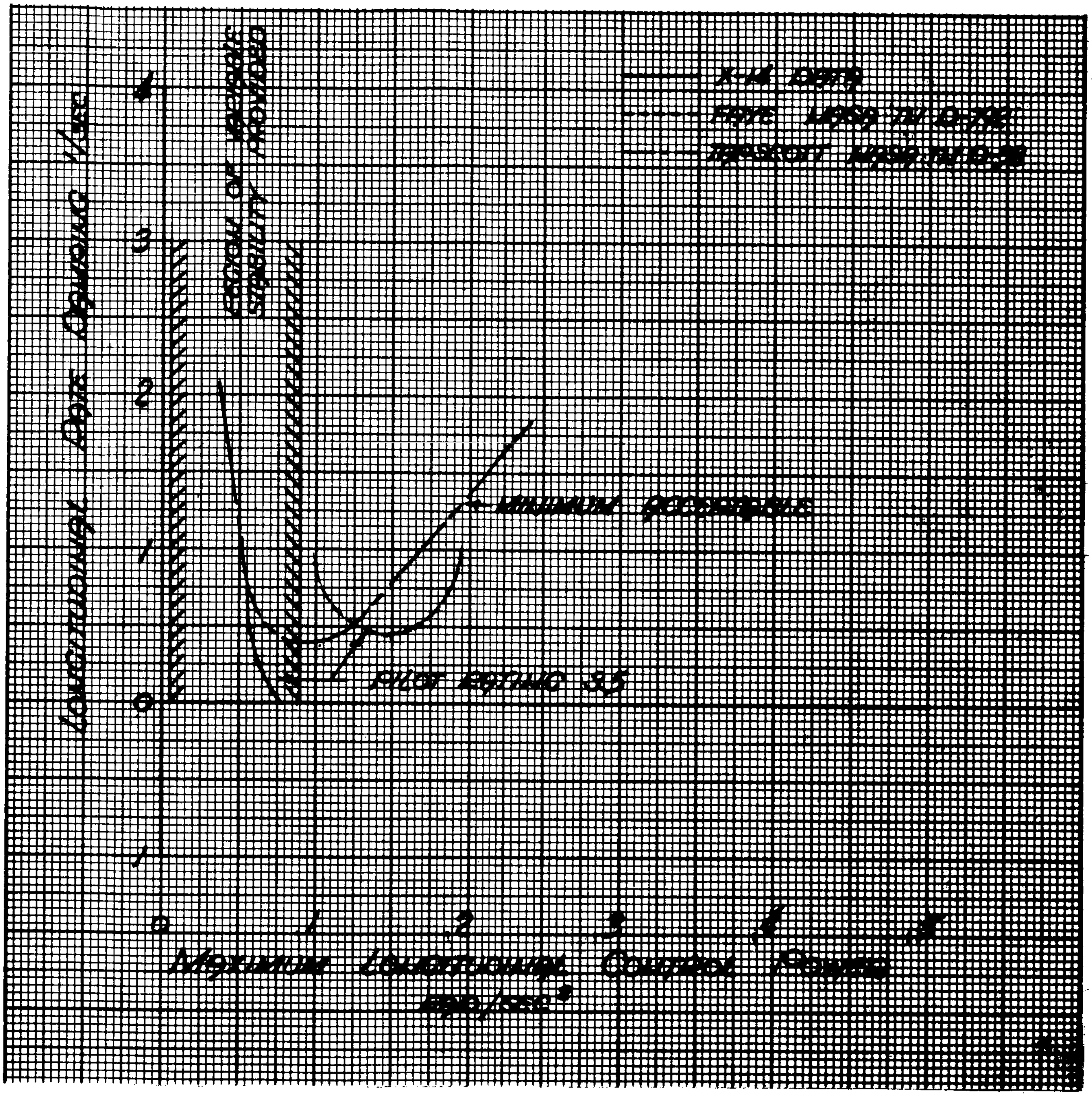

Figure IV-1. Maximum Longitudinal Control Power

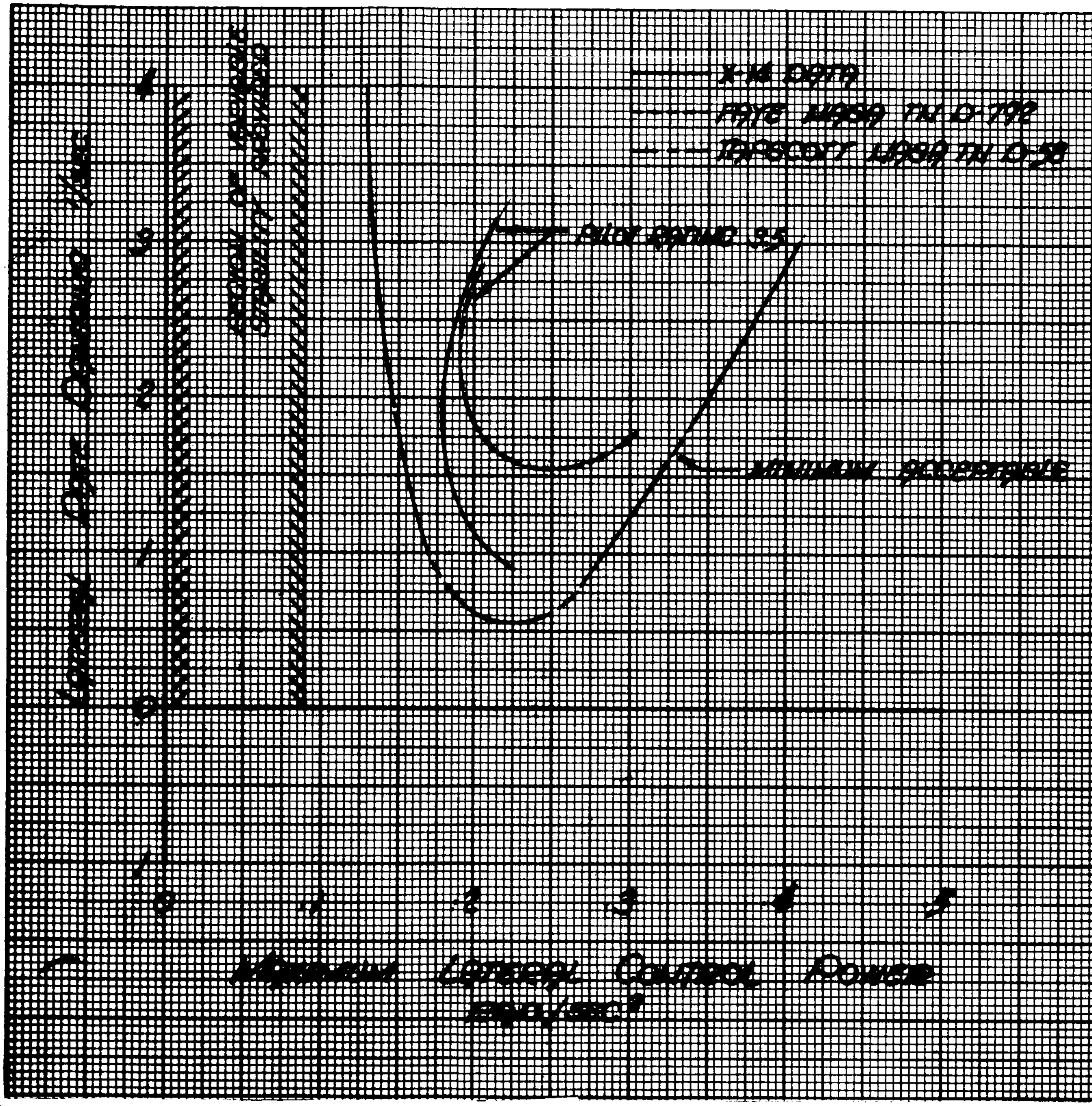

Figure IV-2. Maximum Lateral Control Power

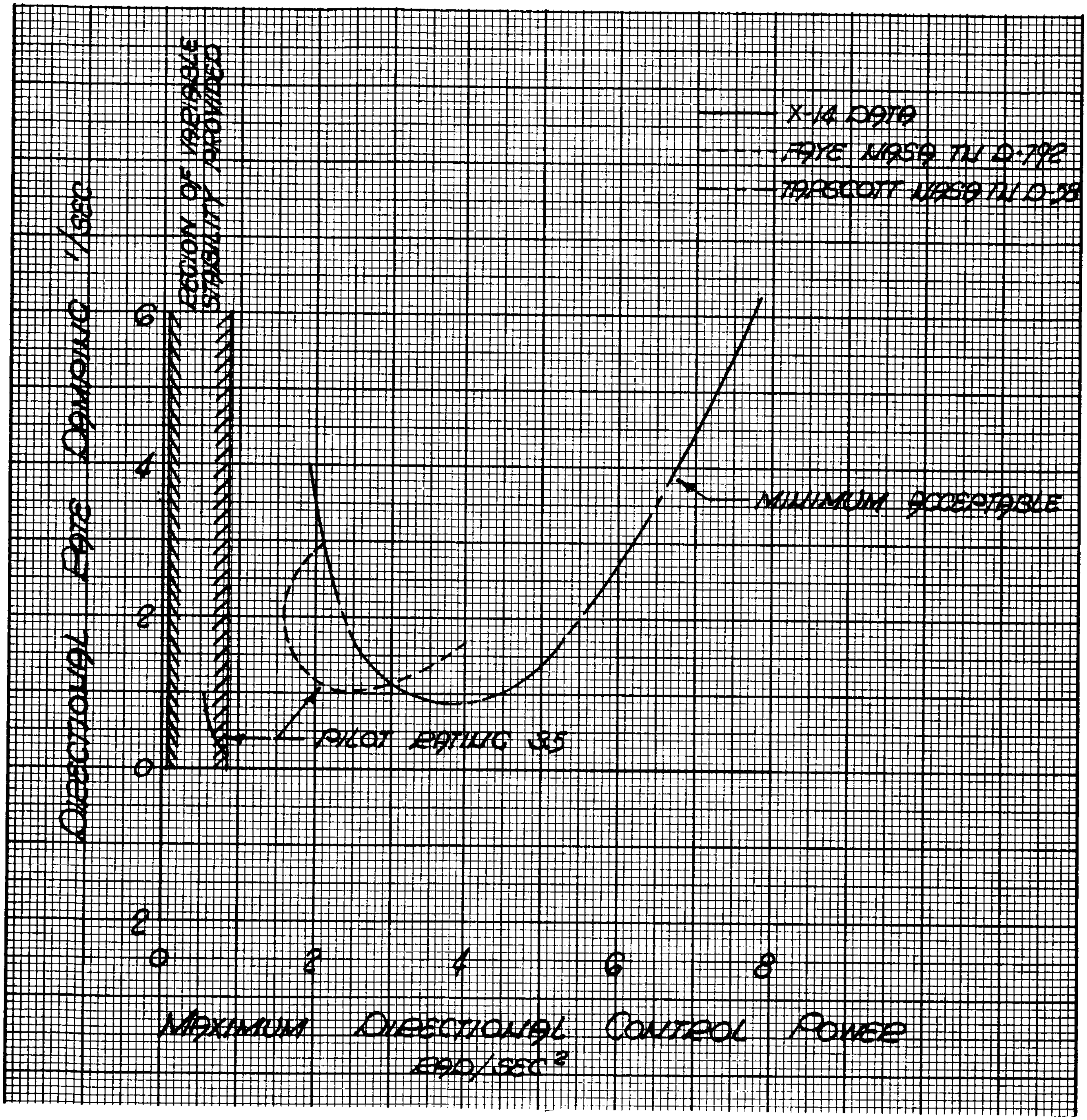

Figure IV-3. Maximum Directional Control Power reduce aerodynamic effects to very small levels (near zero M_u), the X-14 boundaries are considered more applicable to this vehicle because of the low value of M_u inherent in that vehicle. However, another major difference between the VTOL aircraft and the proposed simulator vehicle is that when hovering and maneuvering a VTOL, translational accelerations are more sensitive to attitude change than they would be for the proposed simulator. A given attitude change in a VTOL effectively tilts a 1.0-g thrust vector whereas attitude changes in the simulator effectively tilt only a 1/6-g thrust vector. This effect may explain the lower values of control power found to be optimum for the vehicle from the analog simulation studies conducted on this contract - discussed in detail in Section IV.F.

Based upon these VTOL and helicopter handling qualities studies, it was felt that the variable attitude control power provided in the simulator should cover a range up to approximately 1.0 rad/sec^2 about all three axes. The lower limit of variable control power provided is approximately .05 rad/sec^2. Studies of actual lunar landing vehicles indicate that the actual vehicle will be limited to values of attitude control power of .1 rad/sec^2 or less. Therefore, the variable control power provided in this simulator will cover a range well above and well below this region.

The results of preliminary piloted analog simulation studies for this simulator, conducted during this contract, have indicated that optimum control power levels are near .3 rad/sec^2. These results are presented and discussed in detail in Section IV.F. The range of control power presently selected for the simulator are summarized in Table IV-1.

TABLE IV-1

SUMMARY OF ATTITUDE CONTROL POWER PROVIDED

Control Axis	Moment of Inertia (Slug-Ft^2)	Control Power (M/I) Rad/Sec^2	
		Minimum	Maximum
Pitch	2827 (takeoff) 2551 (landing)	.04 .044	.79 .88
Roll	2474 (takeoff) 2473 (landing)	.045 .046	.905 .915
Yaw	3165 (takeoff) 2662 (landing)	.035 .042	.71 .84

The levels of damping, Mq/I, provided in the control system are practically unlimited as will be discussed subsequently.

The system also must provide the reaction moments needed to counteract the following moments and disturbances:

a.	Jet thrust misalignment with the center of gravity @ nominal thrust	100 lb-ft
b.	Lateral accelerations from engine tilt acting on the vehicle center of gravity	25 lb-ft
c.	Drag moments resulting from center of pressure not being at the center of mass	330 lb-ft
d.	Dynamic unbalance because of asymmetrical configuration	5 lb-ft
e.	Engine tilt torques transmitted through gimbal friction	3 lb-ft
f.	Unbalanced thrust from lift rockets	140 lb-ft

The vehicle attitude control system provides the pilot with the capability of tilting his vehicle ± 30 degrees about the pitch and roll axes and 360 degrees about the yaw axes. Controlling tilt to an accuracy of 1.8 degrees will permit the pilot to control lateral accelerations to a threshold of 0.005-g. To ensure that the pilot does not have to jiggle his control stick at high rates in order to maintain this accuracy, the minimum angular acceleration threshold of the control system should be less than 0.05 radians/second^2. Using the pilot response time of 0.2 second and a dead zone of ± 1.8 degrees, the corresponding limit cycle will be about 5 cycles per minute.

2. Jet Engine Attitude Control

The attitude stabilization system for the jet engine provides the reaction moments needed to counteract the following moments and disturbances on the jet engine:

a.	Thrust misalignment with the center of gravity	17 lb-ft
b.	Lateral vehicle maneuvering accelerations acting on the engine center of gravity (which is below the gimbal center line).	100 lb-ft

c.	Drag moments resulting from air flow into the engine inlet (see Section IV.D).	210 lb-ft
d.	Engine gyroscopic moments.	4 lb-ft
e.	Vehicle maneuvering torques transmitted through gimbal friction.	3 lb-ft
f.	Dynamic unbalance about the two gimbal axes.	2 lb-ft

An additional torque of 30 lb-ft is required to meet the response bandwidth of the control system. Attitude commands which deflect the engine from the vertical to compensate for drag forces on the vehicle can occur at rates up to 10 degrees per second during a turn maneuver. In lateral flight, engine attitude rates to compensate for drag forces will be less than 1.7 degrees per second. During lateral maneuvers the disturbance moments acting on the engine as a result of lateral accelerations acting on the engine center of gravity which is below the gimbal center line can occur at rates corresponding to the pilot's response capability in the operation of the vehicle control system. Using a pilot response time of 0.2 seconds the moment rates required from the engine attitude system will be 500 lb-ft per second. Torque rates required to counteract the drag moments resulting from air flow into the engine inlet will be less than 31 lb-ft per second.

To keep lateral uncertainty accelerations below the pilot's perceptible threshold (0.005-g) (Reference C), the engine should follow attitude commands with an accuracy of 20 minutes of arc. To keep uncertainty accelerations due to angular motions below 0.005-g at the accelerometer location, the limit cycle frequency, for a null uncertainty of 20 minutes of arc in the control system should be less than 3.8 radians per second.

The attitude of the engine should be capable of being deflected ± 10 degrees from the vertical about two perpendicular axes in the horizontal plane to enable it to compensate for drag effects at lateral velocities up to 70 ft per second.

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Reference C - M. F. Marx, Adaptive Controls, Proceedings of the Self-Adaptive Flight Control Systems Symposium; WADC Technical Report 59-49, March 1959.

The drag compensation system provides the means for measuring the drag forces on the vehicle and for providing signals to drive the jet engine attitude and throttle controls to nullify the drag effects on the vehicle.

To obtain a realistic simulation, the effects due to atmospheric drag should be reduced to 0.005-g. A vehicle tilt angle of 1.7° is required for its rocket lift engines to counteract this .005-g drag force. 1.7° is below the threshold which the pilot can sense.

3. Jet Engine Throttle Control

The throttle control system for the jet engine provides the means for automatically positioning the throttle quadrant in response to commands from the vehicle weight programmer and vertical drag sensor.

The weight programmer is capable of commanding a total change in engine thrust of 900 lbs at rates up to 1.7 lbs per second. The vertical drag sensor is capable of commanding a total change in engine thrust of 600 lbs at rates up to 94 lbs per second.

To keep vertical uncertainty accelerations below 0.005-g, the engine throttle control should follow thrust commands with an accuracy of 15 lbs.

A manual throttle is used by the pilot to bring the thrust of the engine up to the 5/6-g value. The automatic throttle control is then engaged and operates the throttle about this nominal setting.

Limit stops and pilot indicators are provided with the throttle servo to prevent the control system from commanding thrust changes in excess of 1500 lbs and to notify the pilot in the event of a failure so that he can disengage the automatic system and operate manually.

The output torque from the throttle servo is sufficient to position the throttle at the maximum rates of command, and yet low enough to enable the pilot to override the automatic throttle servo when he so desires, by approximately doubling his normal force at the throttle. A response bandwidth greater than 4 radians/second is required of the servo.

C. DESCRIPTION OF FLIGHT CONTROL SYSTEMS

A block diagram of the flight control system for the lunar landing simulator is shown in Figure IV-4. It consists of the pilot, his flight indicators, selector panels, control stick, pedals, lift engine throttle quadrant, the vehicle stabilization system, the jet engine stabilization system, and the jet engine throttle system.

In operation, the pilot selects a mode of control and moves his control stick, pedals, and the throttle quadrant in response to what he sees through his window and on his flight instrument panel.

1. Vehicle Stabilization System

The vehicle stabilization system consists of two manually controlled hydrogen peroxide reaction jet systems mechanically and electrically linked to the pilot controls.

The manual control system consists of the pilot control stick and pedals which are connected by linkages to two sets of valves which meter hydrogen peroxide to the pitch, roll and yaw reaction jets. The system also includes the lift rocket throttle quadrant which is connected by linkage to the two valves which meter hydrogen peroxide to the two lift rockets.

A conventional center mounted pilot control stick is used. It commands proportional pitch attitude accelerations when moved fore and aft, and proportional roll attitude accelerations when moved laterally. Motions of the pedals command proportional yaw attitude accelerations.

The manual valve used to meter the hydrogen peroxide to each pair of attitude jets is the same for each axis. The valve consists of a closed center bi-directional spool which meters propellant to either one of the opposed pair of jets. Two sets of paired jets are used for each axis. The system can thereby provide true moments on the vehicle without translation, and in case of failure of one pair of jets, the other could still provide attitude control sufficient to safely land the vehicle.

The lift rocket throttle quadrant is located on the left side of the pilot. The thrust of the lift rockets is proportional to the position of the throttle quadrant.

The manual valve used to meter the hydrogen peroxide to the lift rocket is a larger version of that used for the attitude control system.

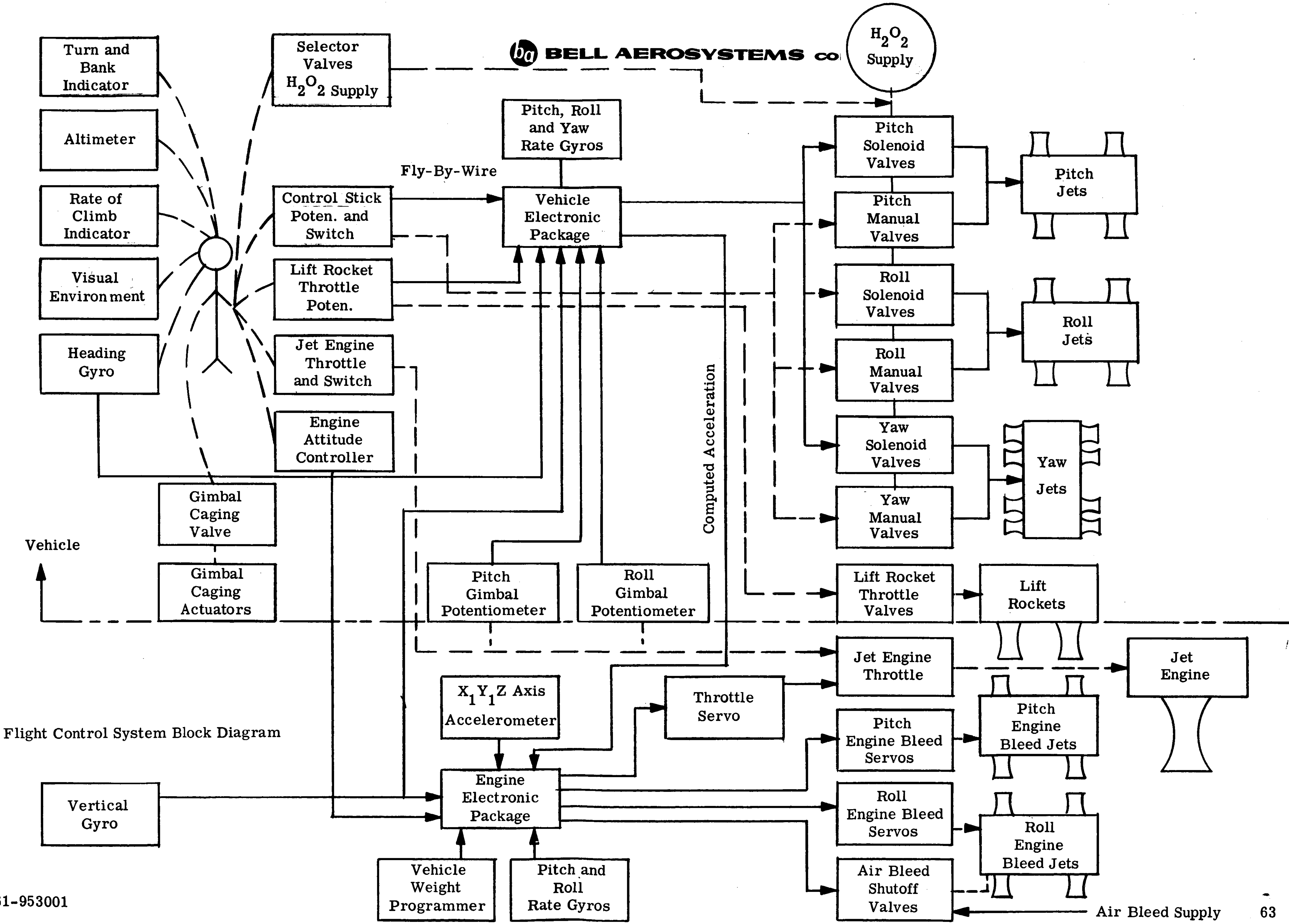


Figure IV-4. Flight Control System Block Diagram

The electrical control system consists of a potentiometer on the control stick which provides an electrical signal to the vehicle electronic package to actuate the solenoid valves which meter hydrogen peroxide to the same pitch, roll and yaw reaction jets as are used in the manual system, i.e., the solenoid control valves are connected in parallel with the manual valves.

On-off solenoid valves are used to meter the fuel to the pitch, roll and yaw jets. To provide a vehicle angular acceleration which is proportional to control stick position, the signal from the control stick potentiometer is used to modulate the width of a one-second pulse in the electronic package. The duration of the pulse determines the angular velocity increment added to the vehicle during each second. The one-second maximum width of the pulse is adjustable to enable the pilot to select a width which is compatible with his response and range of control.

A total of 16 on-off solenoid valves are used, one for each jet nozzle. In addition to being used in the electrical mode of operation, the solenoid valves also receive the signals from the variable stability system.

2. Variable Stability System

The variable stability system consists of the rate gyro and the attitude gyro loops in the vehicle attitude stabilization system. Similar loops are used in each of the pitch, roll and yaw channels, therefore a description of one will suffice for all three.

The rate gyro provides an electrical signal to the electronic package. The analog signal is converted to a time modulated pulse which is used to actuate a hydrogen peroxide valve and jet to impart an increment of angular velocity to the vehicle. A switch in the capsule allows the pilot to choose the sensing of the feedback signal, negative for increased damping and stability, or positive for greater instability. The sensitivity of the feedback signal is made adjustable by a potentiometer on the pilot's selector panel. At zero sensitivity or potentiometer rate gain setting, the attitude control system is essentially an acceleration command system. The sensitivity of this system is adjusted by varying the excitation voltage on the control stick potentiometer. With the rate feedback gain at its maximum negative value the attitude control system corresponds to a velocity command system, and the sensitivity of this system is adjustable by changing the rate gain setting.

The attitude command switch on the pilot's panel permits the pilot to change the sensing of the attitude feedback signals from the attitude gyros and gimbal potentiometers, negative for static stability and positive of static instability simulation. An attitude gain potentiometer on the panel permits the pilot to adjust the attitude loop gain as desired. With the gain set at its maximum value, the attitude commanded is proportional to the position of the control stick and the response bandwidth of the control system is greater than that of the pilot whereas at the very low gain settings the system can be very sluggish to attitude commands and will act more like a velocity or acceleration command system depending on the rate feedback gain setting.

3. Jet Engine Stabilization System

The jet engine stabilization system is shown in in Figure IV-5. It contains engine compressor bleed air jets which are servo controlled to a vertical gyro reference, and to the pilot's engine attitude controller. Electrical signals from the vertical gyro pitch and roll transducers and from the controller are shaped in the engine electronic package to drive an electric actuator which controls the bleed air nozzle openings.

The vertical gyro is mounted to the engine supporting structure. Four orthogonal structural extensions located in a horizontal plane support the constant bleed air nozzles and provide the desired torque arms. Two nozzles are used for pitch and another two for the roll axis. This arrangement maintains a constant demand on the bleed air supply thereby minimizing thrust variations in the jet engine.

The engine electronic package is designed to also receive signals from the lift rocket throttle position potentiometer, the vehicle attitude gyros and potentiometers and the drag compensation system. The inertial acceleration of the vehicle is computed from the attitude and throttle position signals. X, y, and z accelerometer signals are subtracted from the computed inertial acceleration to determine vehicle drag. The computed drag signal along the x and y axes are supplied to the pitch and roll axes bleed air nozzle servos respectively to tilt the engine in a direction to nullify the drag signal. The z-axis signal is supplied to the jet engine throttle servo in a direction to nullify the drag signal.

Two rate gyros are mounted to the engine structure to measure pitch and roll rates. These signals are needed to damp the jet engine attitude stabilization loops in the linear operating ranges of the system and to constrain the limit cycle in the null or dead zone range of the control system.

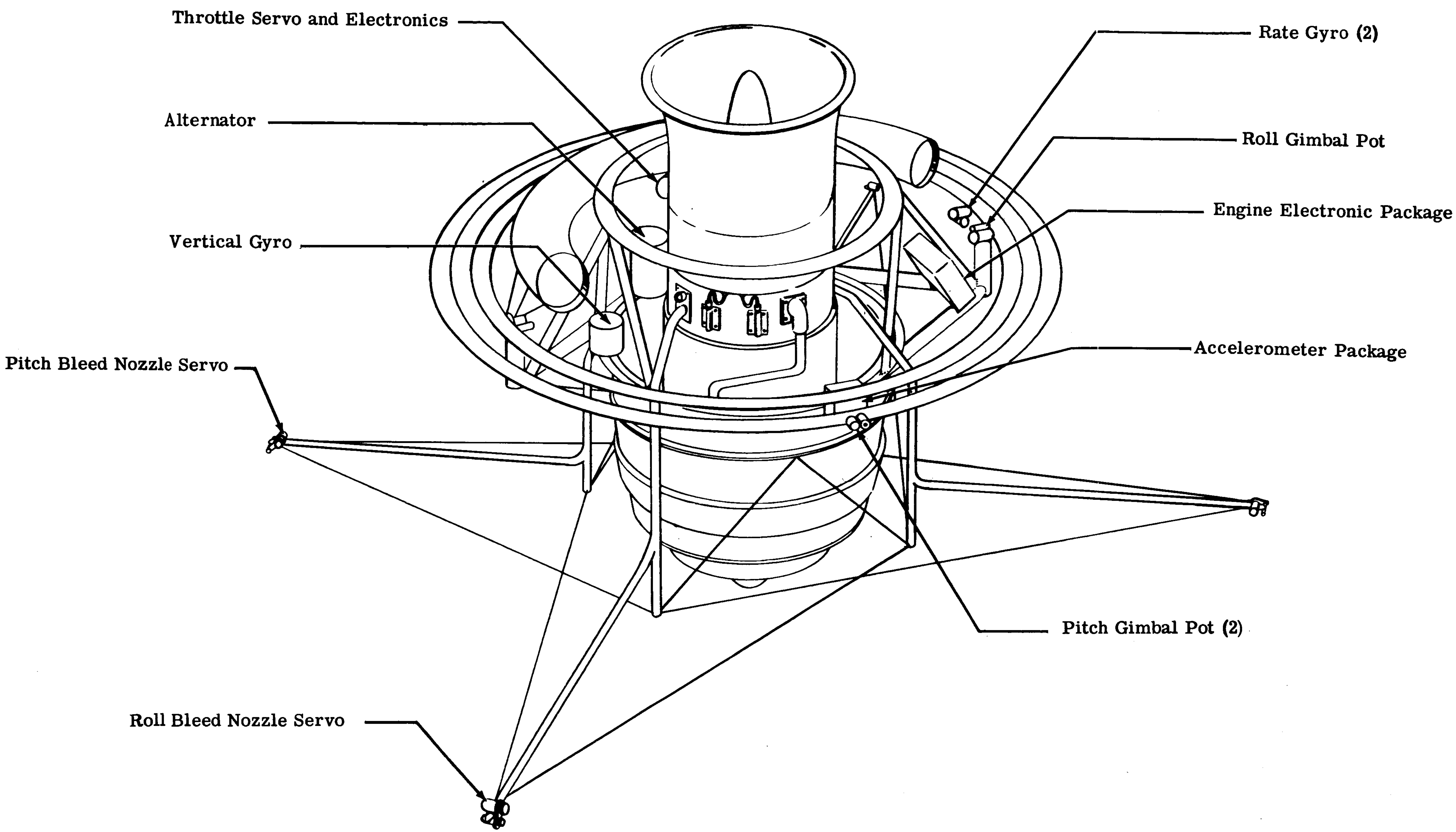


Figure IV-5. Location of Engine Stabilization Components

A free gyro will be used for the jet engine and vehicle vertical reference. Several low cost mass produced, flight qualified autopilot gyros are available which can be erected by a bubble pendulus reference prior to takeoff to an accuracy of better than 1/4 degree. A 1/4 degree error in jet engine attitude would be automatically compensated by the pilot, during simulated lunar flight, by tilting the vehicle approximately 1-1/4 degree in the opposite direction. This is below the threshold which the pilot can detect. With the erection circuit disconnected, the drift rate of a typical autopilot gyro will be 0.1 degree per minute, which would allow a reference error of 1.0 degree in a ten minute flight. Since this error cannot be tolerated, it would be necessary to maintain continuous gravity vertical erection during flight, as is standard practice in aircraft and helicopter autopilots.

The simulation can be improved by using a very low drift rate free gyro such as the Minneapolis-Honeywell GG-87. This is a miniature integrating gyro particularly well suited to strapdown guidance applications. The present design has an input freedom of ± 10 degrees, although ± 60 degrees is possible. This gyro will have a drift rate of one to two degrees per hour without trimming. By trimming before flight the drift rate can be reduced to about 1/10 degree per hour.

The cost of this gyro is an order of magnitude greater than the conventional autopilot gyro, and the gyro must be temperature stabilized.

Further analysis is required to determine whether an aircraft-type autopilot or a low drift free gyro should be used in this application.

A stabilized platform, mounting three orthogonal gyros could, of course, provide excellent attitude information. However, the cost and delivery schedule for presently available platforms, which weigh less than 100 lbs, would not be compatible with early and low cost vehicle delivery.

The drag compensation system computes the accelerations which would result in the absence of drag, compares this with measured actual lateral accelerations, and applies the difference as a command to tilt the jet engine and/or drive the engine throttle to reduce the difference to zero.

The product of the signals from the lift rocket throttle position potentiometer and the attitude sensors determines the commanded acceleration. Instrument servos are used to provide this computation. Three orthogonally mounted accelerometers measure the x, y and z components of acceleration. The signal from the accelerometers is subtracted from the computed signals and the difference is used to drive the jet engine throttle and bleed air nozzle servos.

The thrust from the lift rocket engines is repeatable within 5% with throttle setting, and for the operating range, the attitude of the vehicle is measured within 2%. These errors result in a computed acceleration uncertainty of 0.009-g. The measured acceleration has an uncertainty of 0.01-g due to the threshold of the accelerometers.

Another scheme considered for the drag compensation system involved the use of pressure probes. This scheme requires the use of multiple fixed heads or one weathervaning head. Pressure transducers for this low range of pressure are not readily available for this type of application, and the use of anemometers results in questionable accuracy if the anemometers are fixed, and increased complexities if the anemometers are positioned into the velocity vector.

The third method of providing compensation for the atmospheric effects is to use a computer to determine the required additional thrust. In this method, the acceleration to be expected under lunar conditions for any throttle setting is determiend beforehand. With this acceleration value known for a throttle setting, the added thrust required to overcome the atmospheric drag is computed as a function of the time during which the throttle setting is maintained.

The calculation of the required drag compensation as a function of the throttle settings is probably the least complex of the methods considered. However, this method will fail to perform realistically if the performance of any of the thrust devices deviates from that obtained in the calibration of the throttle settings, and the error will be integrated and increase with time.

The accelerometer method has been tentatively selected, but a search for a suitable weathervaning probe will continue.

4. Jet Engine Throttle Control System

The basic block diagram for the thrust control system is shown in Figure IV-6. The command signals to the servo system originate from the aerodynamic drag computer and the weight programmer. These signals are used to actuate a position servo system which moves the jet engine fuel control linkage in response to the commands. The fuel control linkage is mechanically coupled to the engine and the throttle quadrant so that both are moved in response to position changes of the servo output shaft.

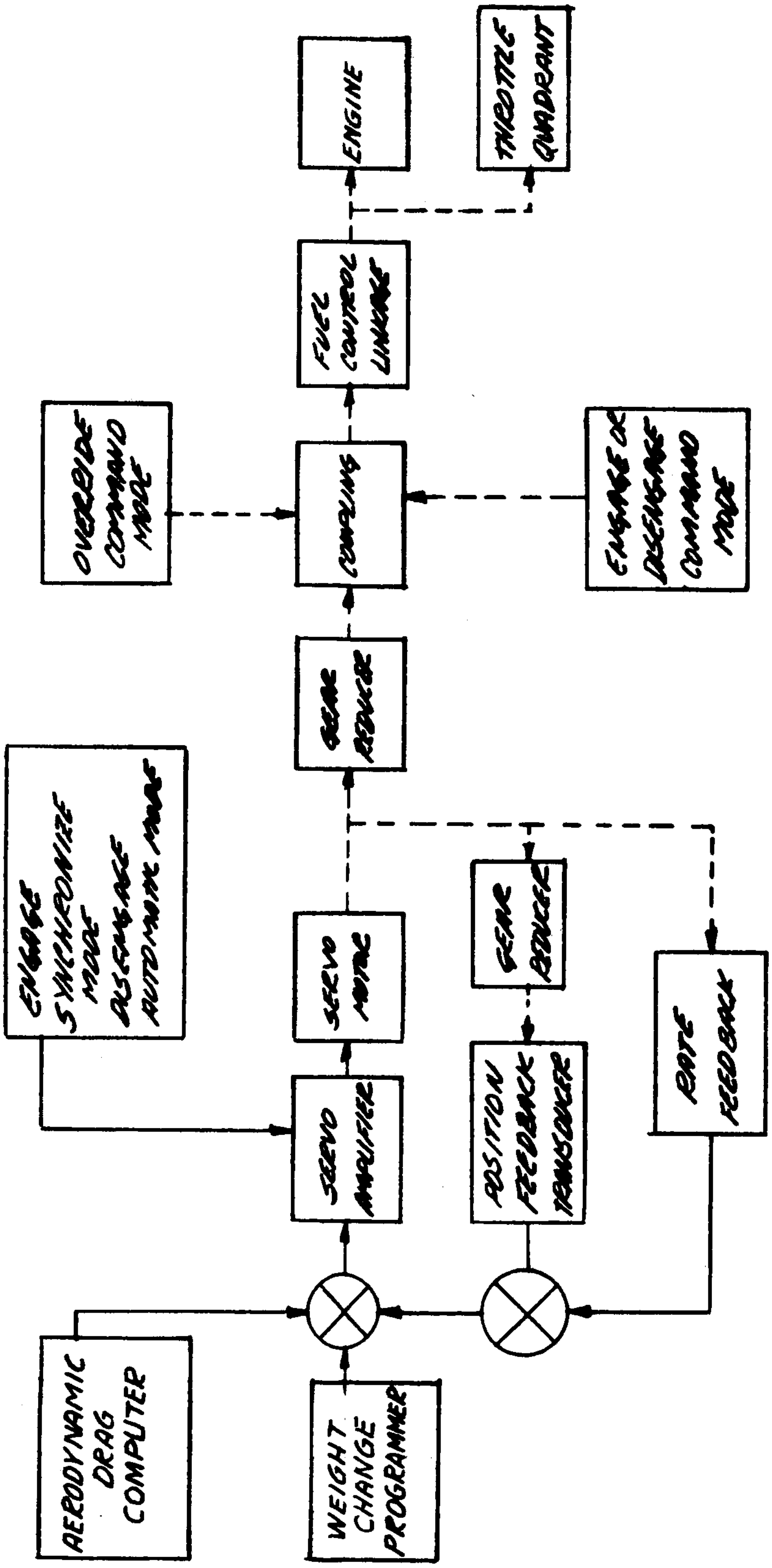


Figure IV-6. Jet Throttle Control Block Diagram

The system is designed to have two complementary operating modes, the synchronizing mode and the command mode.

a. Synchronizing Mode

With the synchronizing mode engaged, the pilot sets the jet engine throttle manually to the desired level, using the throttle quadrant and thrust indicator. In addition to adjusting the engine throttle, the throttle input signals are entering the control system and are being established as reference values. Since the coupling between the engine linkage and the servo motor shaft is not engaged at this time there is no coupling between the servo and the fuel control linkage. This also reduces the load on the servo motor and synchronization is accomplished with minimum elapsed time since the motor is running at maximum speed. When synchronization is complete the servo motor is stopped and the existing inputs are established as reference values. When the command mode is engaged there will be no transients to the fuel control linkage and therefore no undesirable vehicle maneuvers.

b. Command Mode

Engaging the command mode is effected by coupling the servo motor to the fuel control linkage. In this mode the fuel control linkage position, and therefore engine thrust, is changed automatically to properly compensate for vehicle weight change and aerodynamic drag change. The system is designed in such a manner that the pilot can readily overpower the command mode in the event of a malfunction or if it is necessary to make a gross power change. Manual override is accomplished by decoupling the servo shaft from the fuel linkage without de-energizing the system. The system will therefore continue to operate in the synchronize mode and will be ready if the pilot desires to re-engage it. However, the pilot as well as system fail safety interlocks will have the capability to completely disengage the system.

5. Electronic Subsystem

The electronic subsystem comprises the Vehicle Electronics Assembly (VEA) and the Engine Electronics Assembly (EEA). The electronic subsystem converts the attitude and throttle signals from the electromechanical transducers to suitable signals for use by the control system's servos.

a. The Vehicle Electronic Assembly

The Vehicle Electronic Assembly (VEA) is used to combine, amplify and convert the attitude command signals for introduction into the reaction control servos to position and stabilize the vehicle in pitch, roll and yaw.

Consequently, the VEA can be described as housing three identical sets of electronics, one for each of the three attitude control servos. Each channel consists of two A.C. signal amplifiers, two demodulators, one summing amplifier, and an analog to digital converter (Figure IV-7).

The operation of each channel can be briefly described as follows (Figure IV-8):

When the pilot issues a command to change the vehicle's attitude, this is transmitted electrically by means of an infinite resolution potentiometer, energized by a D.C. voltage, to the VEA. This command is then transmitted through a summing amplifier to the analog to digital (A-D) converter where it is converted to a pulse width modulated signal and applied to the reaction jet servo valves.

The momentum imparted to the vehicle by the reaction jets causes an angular change in the vehicle position about this axis which is sensed by the attitude gyros. This position attitude change is reflected by a change in the A.C. signal level of one of the gyro synchro pick-off axis. This electrical signal is amplified and then converted to a D.C. signal in a demodulator. The demodulator output representing the position feedback signal is added algebraically to the command signal in the summing amplifier. To insure stability of the servo loop, a rate gyro signal proportional to the rate of change of the vehicle's position is also added in the summing amplifier. Because the rate signal from the gyro is also an A.C. signal it is first amplified and then demodulated.

The on-off operation of the reaction jets requires a device which will convert an analog control signal into a pulse, the duration and phase of which is proportional to the magnitude and phase of the signal.

The schematic for this device is shown in Figure IV-9. The input signal from the summing amplifier is coupled to a bridge rectifier to produce a negative signal. This negative signal is summed with a positive saw tooth and applied to the base of Q_1. The saw tooth swing is from +1 to +28 volts so that with zero volts out of the bridge rectifier Q_1 is always in conduction and the collector voltage is approximately zero. As the negative voltage at the output of the bridge rectifier increases Q_1 is cut off for a portion of the saw tooth cycle, producing positive output pulses. The saw tooth period will be one second duration and the output pulse width will vary from 20 milliseconds to 950 milliseconds.

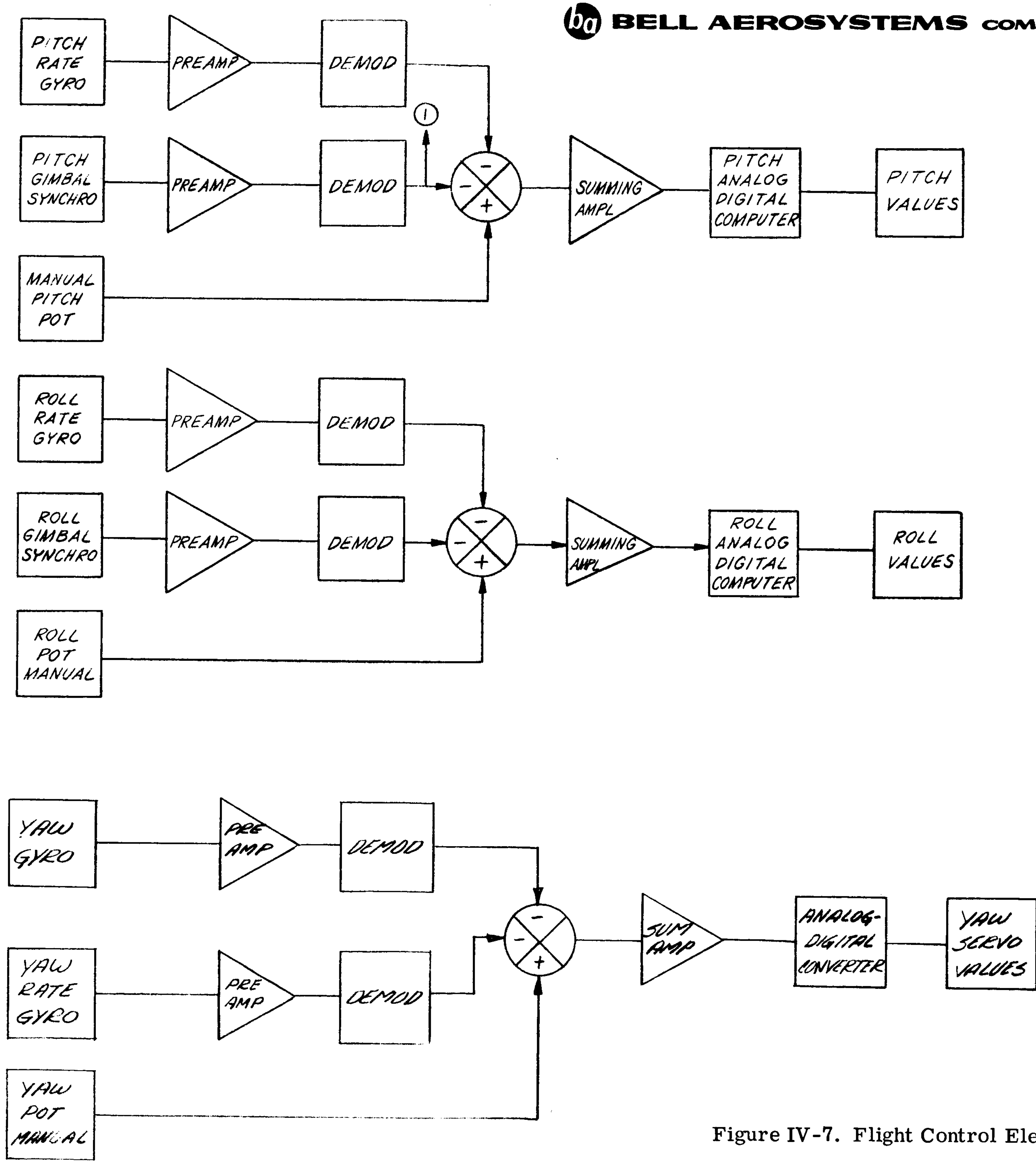


Figure IV-7. Flight Control Electronics Block Diagram

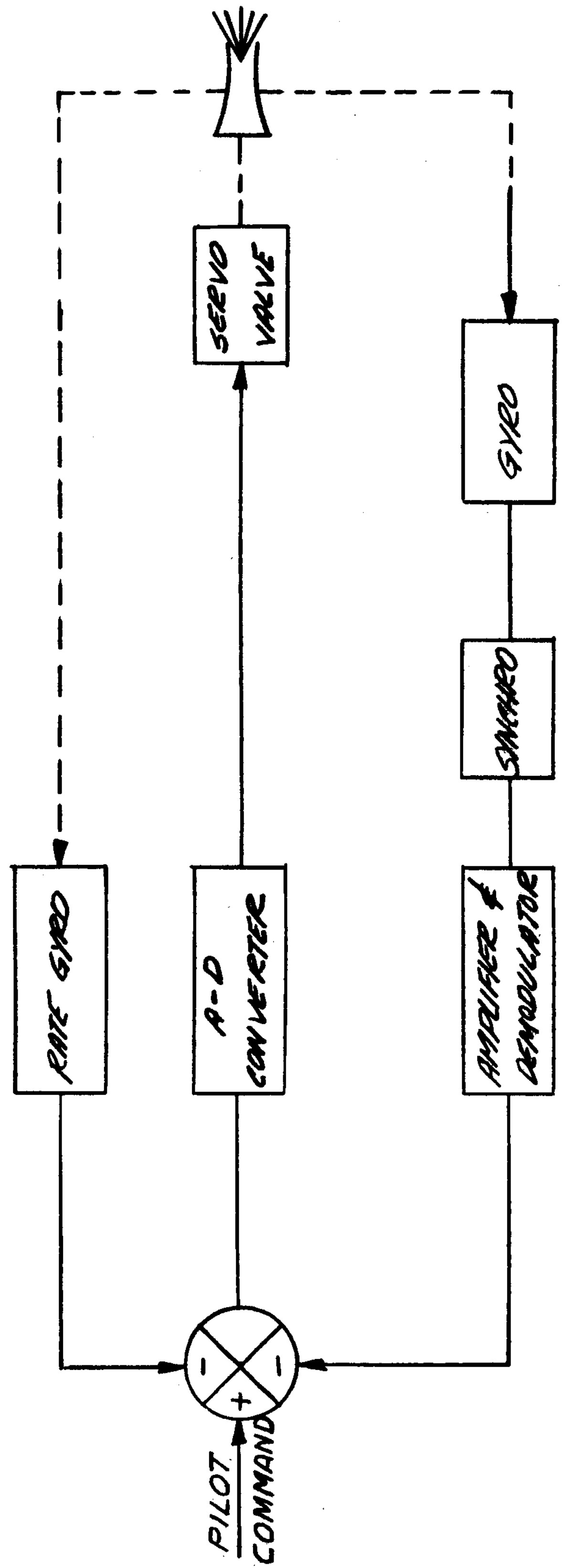


Figure IV-8. Vehicle Attitude Control Loop

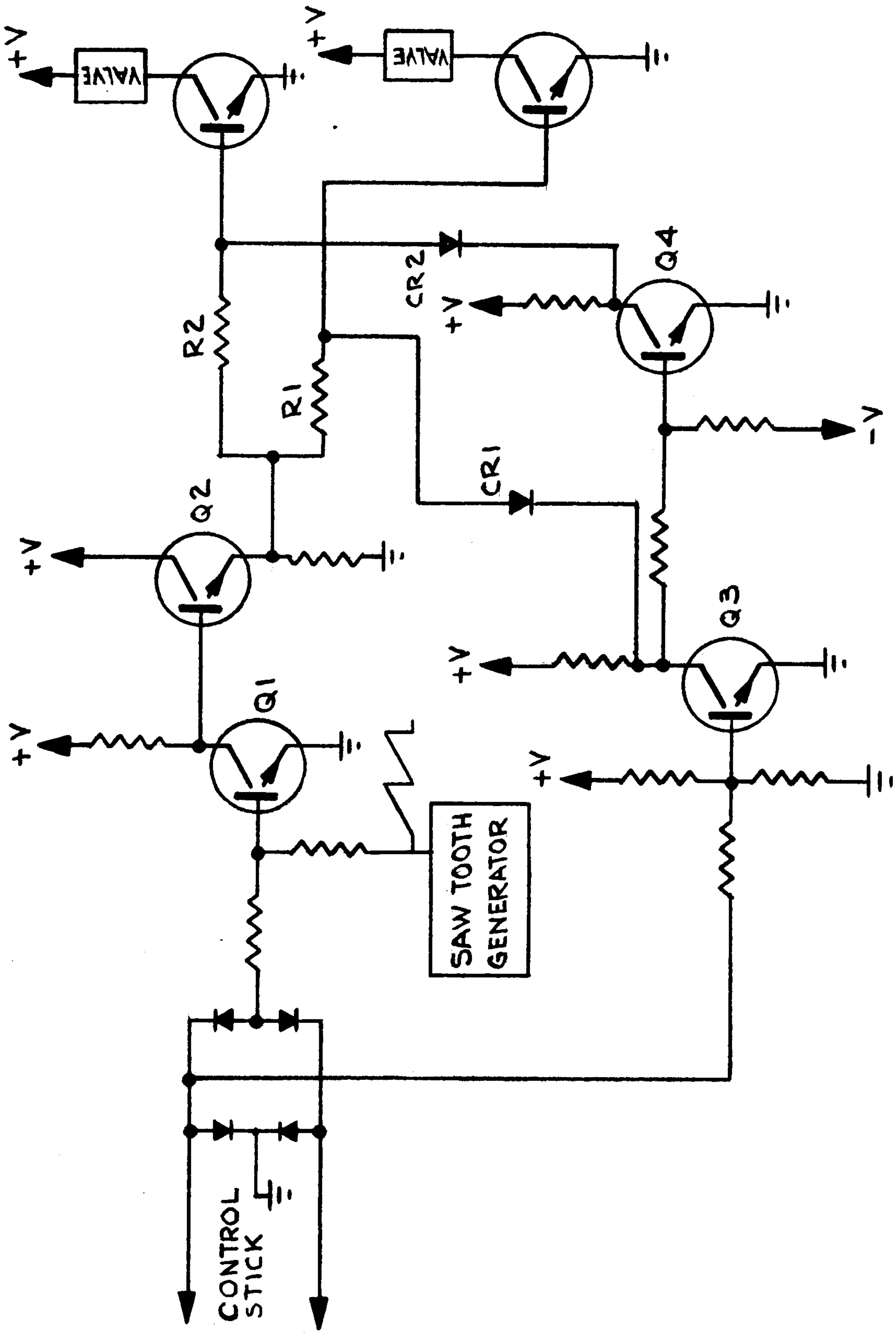


Figure IV-9. Analog to Pulse Converter Schematic

The output from the pulse width modulator is coupled to an emitter follower Q_2. Two outputs are fed from Q_2 through R_1 and R_2. These outputs are controlled by the action of Q_3 and Q_4 through C_{R1} and C_{R2}. With the input line positive Q_3 is conducting and Q_4 cutoff, hence, the output end of R_1 is grounded through C_{R1} and Q_3 and no pulse appears, while with Q_4 cutoff C_{R2} is in a nonconducting state and an output appears at R_2. With the input line negative Q_3 is cutoff and Q_4 conducting causing an output to appear at R_1. The outputs of R_1 and R_2 are power amplified and fed to their respective servo valves.

b. The Engine Electronic Assembly

The Engine Electronic Assembly provides the means for amplifying, combining and converting the gyro and accelerometer signals transducing the jet engine attitude, motion, and thrust, and applying them to the appropriate servos.

The Engine Electronics Assembly is comprised of three channels, pitch, roll and thrust control. The pitch and roll channels are identical in their configuration and component parts. Each channel consists of 2 A.C. amplifiers, 2 demodulators, 1 summing amplifier, 1 drag computer servo, and 2 bleed valve servos (Figure Iv-10).

The jet engine's attitude is measured by a vertical gyro and converted to an A.C. signal by its pick-off. This signal is then amplified and demodulated, thus converting the low level A.C. signal into a high level D.C. signal. It is then transmitted through a summing amplifier to a bleed servo, which provides the necessary power to operate the engine bleed valves causing the repositioning of the jet engine's attitude.

The attitude gyro signal is corrected by a D.C. signal from the drag computer servo, an accelerometer D.C. signal and a D.C. rate gyro signal. These signals are all summed algebraically by the summing amplifier. Since the rate gyro pick-off signal is A.C., it is first A.C. amplified and then demodulated before it is summed with the other signals.

The computed drag signal is derived from an electomechanical servo which positions an infinite resolution potentiometer as a function of rocket engine thrust. The electrical input to the potentiometer represents the engine attitude angle; thus the output of the potentiometer represents the product of the thrust and the engine attitude, i.e.

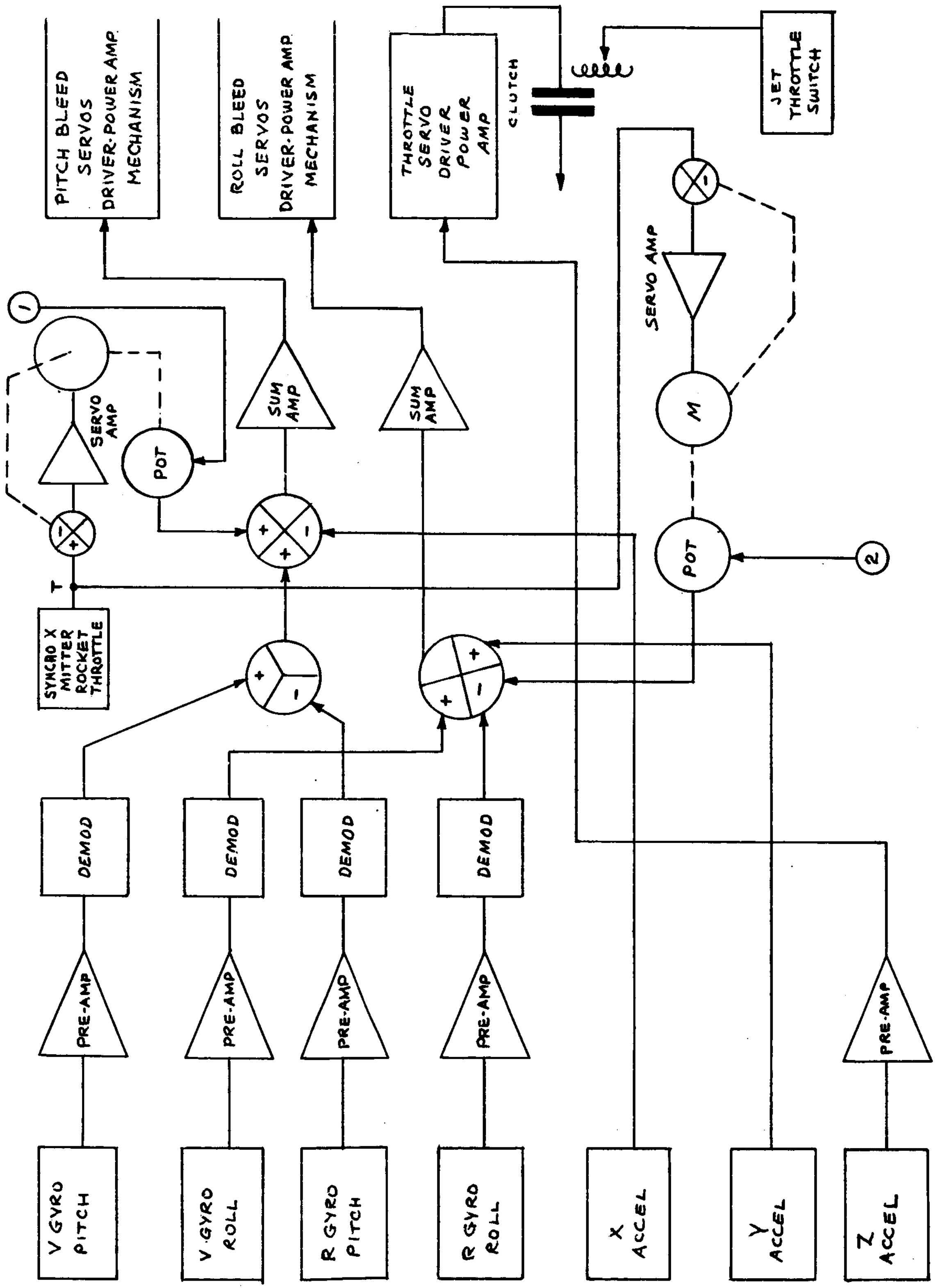


Figure IV-10. Engine Electronics

inertial acceleration. The servo is packaged as a separate plug-in module to facilitate maintenace and replacement. Because the drag computation must be performed as a function of pitch and roll, two such servos are required.

The jet engines thrust is controlled by a throttle servo which accepts an amplified A.C. signal from the accelerometer.

c. The Electronic Circuitry

The electronic circuitry is all solid state (demodulator excepted) and utilizes silicon semiconductors exclusively. As a consequence the power consumption is kept low and reliability is high. The order with which the electronic operations of amplification, demodulation and summing are performed insures a temperature stable, low noise operation with a minimum of circuitry. The ruggedness of the circuit design makes it insensitive to variations in power supply voltages and requires no component selection.

The amplifiers and demodulators are identical to those used in the Vehicle Electronic Assembly, and except for a gain setting, are directly interchangeable.

d. Packaging

All electronic circuitry, with the exception of the power supplies and the servo is mounted on printed circuit board to facilitate interchangeability and maintainability.

The printed circuit cards are guided into their receptacles by spring loaded guide rails which assure positive contact of the connectors and prevent vibration interference.

The Engine Electronics Assembly is mounted to the engine structure and is therefore vibration isolated. The EEA weighs 15 lbs excluding the power servos and is 6" x 8" x 8" in size. Its power consumption is approximately 10 watts and, therefore, no special cooling provisions are incorporated. This unit is sealed against moisture.

The Vehicle Electronics Assembly will be 6" x 8" x 8" in size and will weigh approximately 10 lbs. The power consumption of this unit will be approximately 30 watts. This heat will be dissipated via conversion. The combined power consumption of the gyros accelerometers and power servos is estimated at 150 watts.

e. The Preamplifier

A schematic of the proposed preamplifier is shown in Figure IV-11. It is composed of three stages of amplification and a buffer stage. The specifications for the amplifier are as follows:

Gain:	2000 adjustable 5:1 ratio
Gain Stability:	±5%
Linearity:	1%
Bandwidth:	300 - 500 cps
Input Impedance	50K ohms
Output Impedance:	500 ohms
Noise:	2.5 volts referred to input
Saturation:	10 volts

f. The Demodulator

A schematic of the proposed demodulator is shown in Figure IV-12. It is composed of a transformer, an RC network for phase adjustment and an electromagnetic chopper. This circuit is perferred for this application because drift and noise which are inherent to solid state demodulators are thus circumvented. The reliability of choppers is such that they are not expected to adversely affect the design.

The demodulator is designed to meet the following specifications:

Null Offset:	10 volts
Noise:	100 volts
Linearity:	1%
Saturation:	10 volts
Power:	6.3 volt A.C. - 400~ 30 ma
Gain:	0.5
Phase Shift:	5^{o}

g. The Summing Amplifier

A schematic for the proposed summing amplifier is shown in Figure IV-13. Diode temperature stabilization is used, and this is supplemented by feedback providing a high input impedance.

Input Impedance:	100K
Gain:	1
Gain Stability:	±5%
Saturation:	10 volts
Output Impedance:	2K ohms

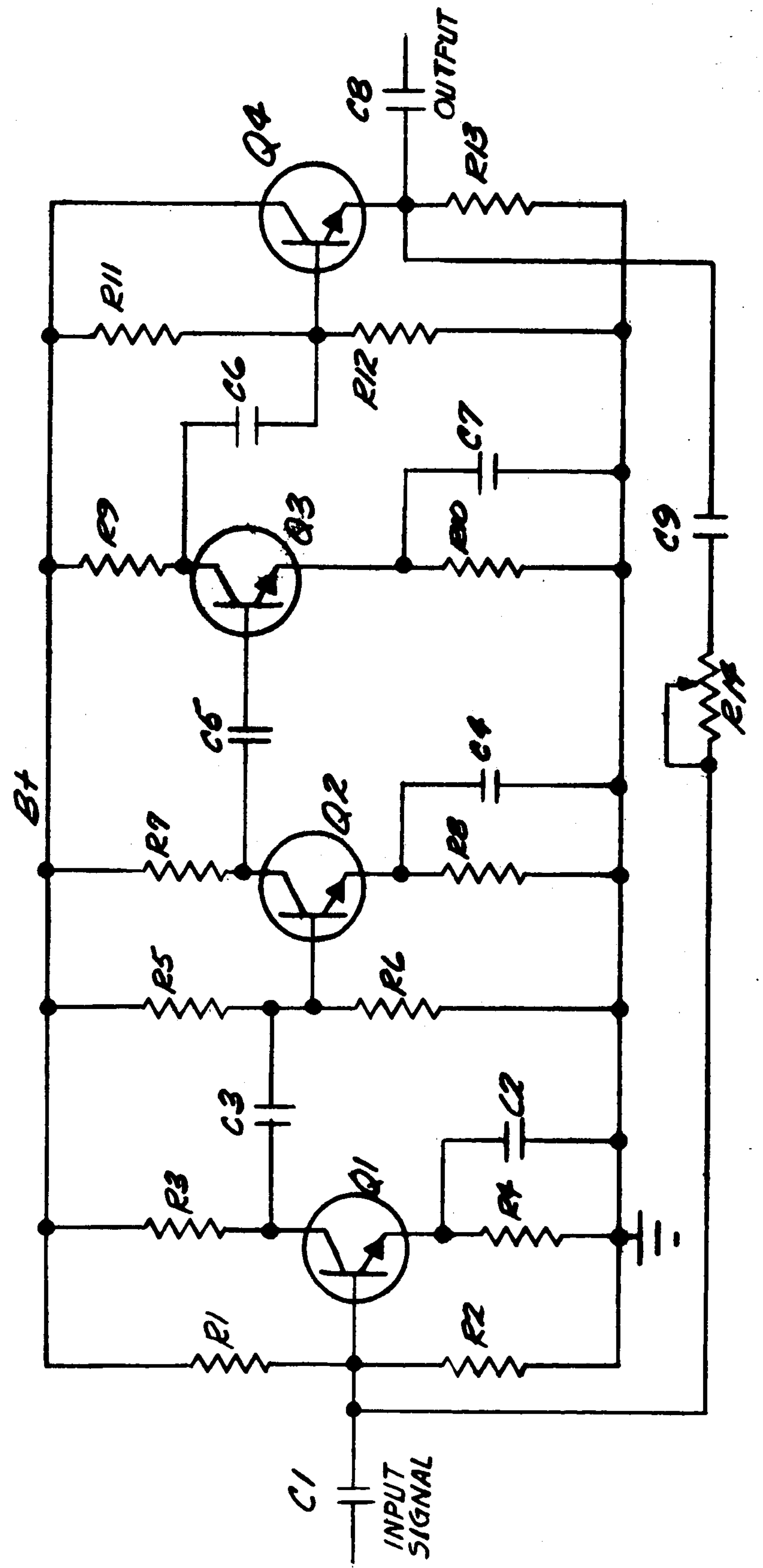


Figure IV-11. Preamplifier Schematic

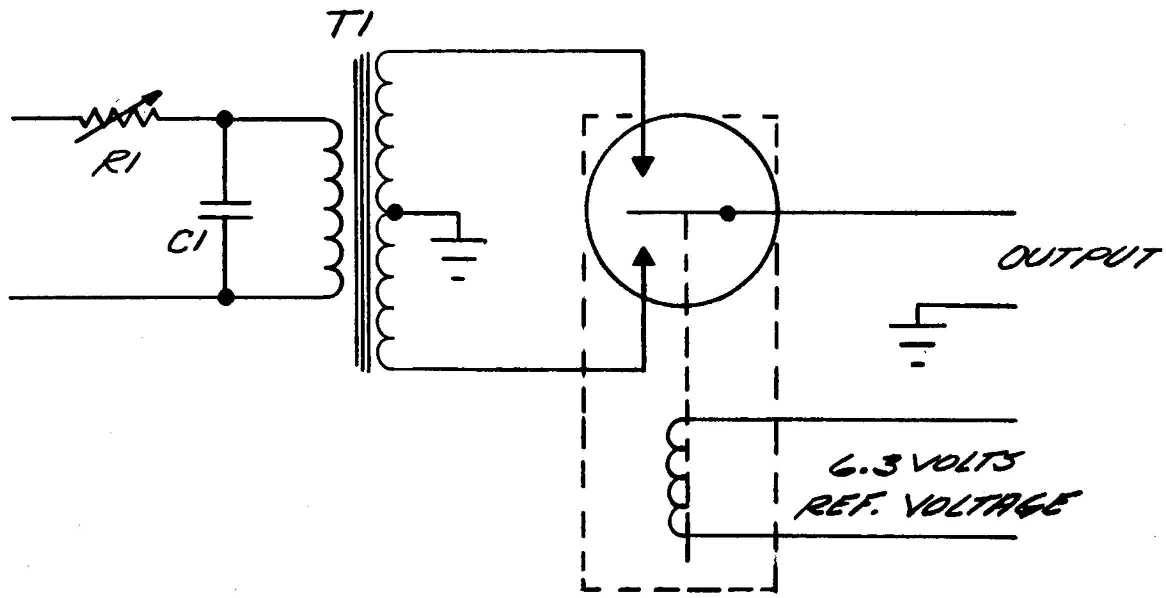


Figure IV-12. Demodulator Schematic

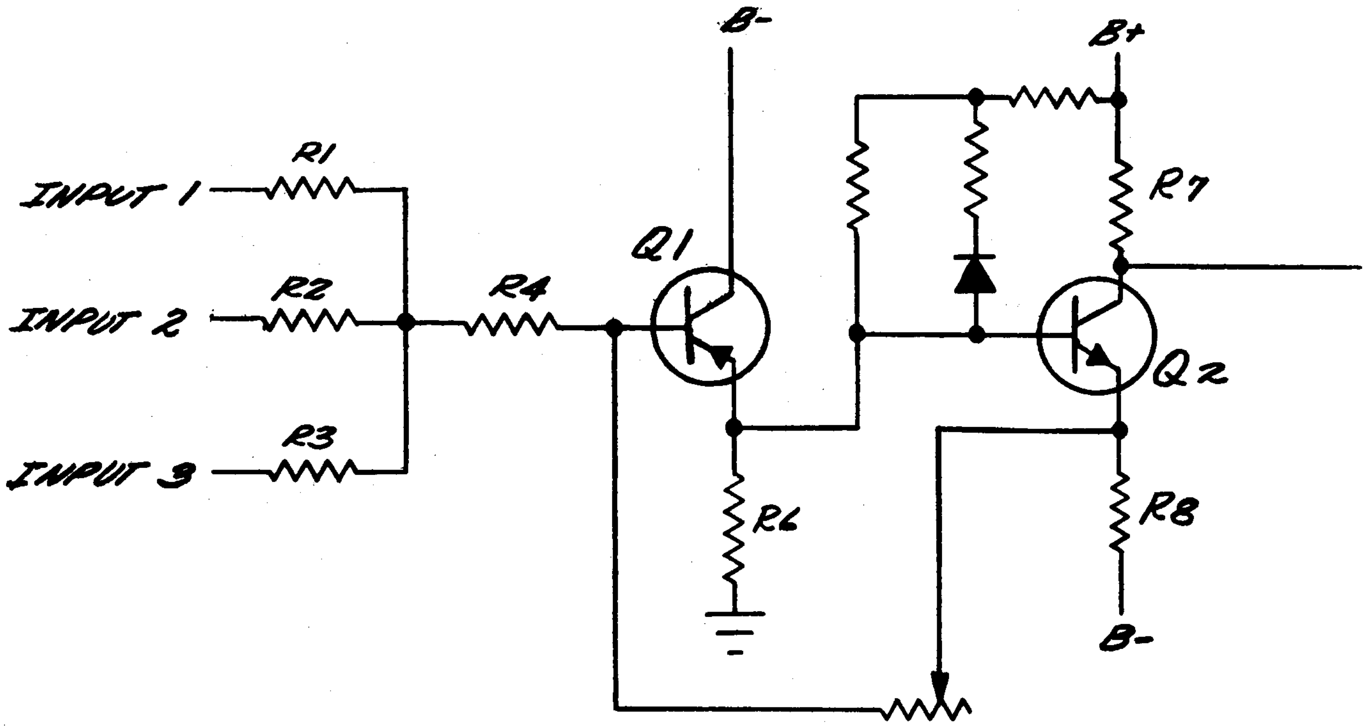


Figure IV-13. Summing Amplifier Schematic

h. Power Supply

The power supplies (Figure IV-14) are +30 and -30 volt D.C. and require only a zener diode for regulation. The following specifications apply:

Input voltage:	115 volt	400 ~ A.C.
Output voltage:	+30 volts	$\pm 10\%$
	-30 volts	$\pm 10\%$
Ripple voltage:	1%	

D. ANALYSIS OF AERODYNAMIC DRAG AND MOMENT CHARACTERISTICS

1. Basic Vehicle Drag, Lift and Pitching Moments

Calculation of the aerodynamic characteristics of the proposed configuration is not a simple analytical process. To facilitate the necessary analyses, some basic assumptions were made. These enabled a reasonable estimate of the aerodynamic parameters of this unconventional type aircraft.

For the purpose of analyzing aerodynamic lift, drag, and moments, the cross flow drag theory was employed to evaluate the drag and lift coefficients of all the circular cylinders that are the major basic components of this configuration. At an angle of attack, α , the flow pattern and dynamic pressure forces on these cylinders correspond to the velocity component in the direction normal to their axes.

Thus, $C_N = \dfrac{N}{qS_{ref}} = C_{D_c} \sin^2\alpha$

where C_{D_c} = cross flow drag coefficient of a circular cylinder.

This normal force is then split into its drag and lift components

$C_D = C_{D_c} \sin^3\alpha$; $C_D = 1.1 \sin^3\alpha + .02$ (including friction drag)

$C_L = C_{D_c} \sin^2\alpha \cos\alpha$; $C_L = 1.1 \sin^2\alpha \cos\alpha$

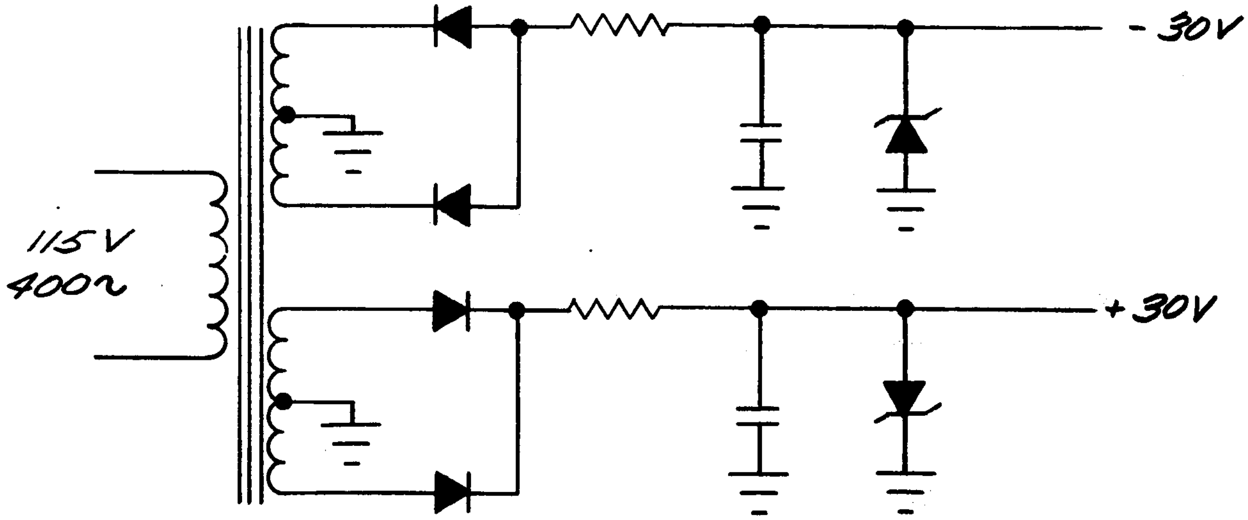


Figure IV-14. Power Supply Schematic

Experimental results on wires, cables, and cylinders (at subcritical Reynolds Numbers) confirm the prediction very well after adding the frictional component. To simplify the theoretical analysis of the aerodynamic characteristics, it was assumed that there is no interference drag. This type of drag is present when two or more bodies are placed one behind the other, or in close proximity to each other. Strict calculation of interference drag would be complicated and specific methods to solve such problems would be quite involved. Although interference drag is neglected, the method used is expected to give a reasonable estimate of the forces and moments acting on the vehicle. It is apparent however, that for this type of configuration an accurate determination of all the aerodynamic forces can best be obtained from wind tunnel testing.

Utilizing the drag and lift equations previously discussed, an IBM program was initiated which provides a rapid and convenient method for calculating and tabulating the aerodynamic characteristics for the complete structural portion of the vehicle. This data is obtained for varying angles of attack, α , and yaw angles, β . The program tabulates total axial, side and normal force coefficients along with total pitch, yaw, and roll moment coefficients. These coefficients are obtained for body, stability and wind axes. The program is sufficiently flexible so that any structural change in the vehicle requires only the new x, y, and z coördinates for each end of the new or altered member. With this new information added to the program, revised force and moment data for the complete vehicle is quickly available. This affords rapid aero-analysis of structural changes to be made with relative ease since a complete analysis of the configuration can be made in a relatively short period of time.

Drag estimates calculated for the capsule by the method just described were added to the drag estimates for the engine (discussed in a subsequent section), and the total drag is shown in Figure IV-15 as a function of velocity and angle of attack, α . Figure IV-16 shows that there is no significant effect of sideslip angle on the vehicle body axis drag. Figure IV-17 shows the calculated vehicle normal force as a function of velocity and angle of attack. Several factors, which due to their complexity in calculating, have not been included in these estimates. These would be interference drag and drag due to the gusset plates used at the terminal junctions of the cylinders. The causes and effects of this drag were previously discussed. The effect is best determined from experimental test data.

The method of drag compensation is to command a jet engine tilt angle in the direction of motion, based upon error signals between command and actual acceleration. The magnitude of tilt required is

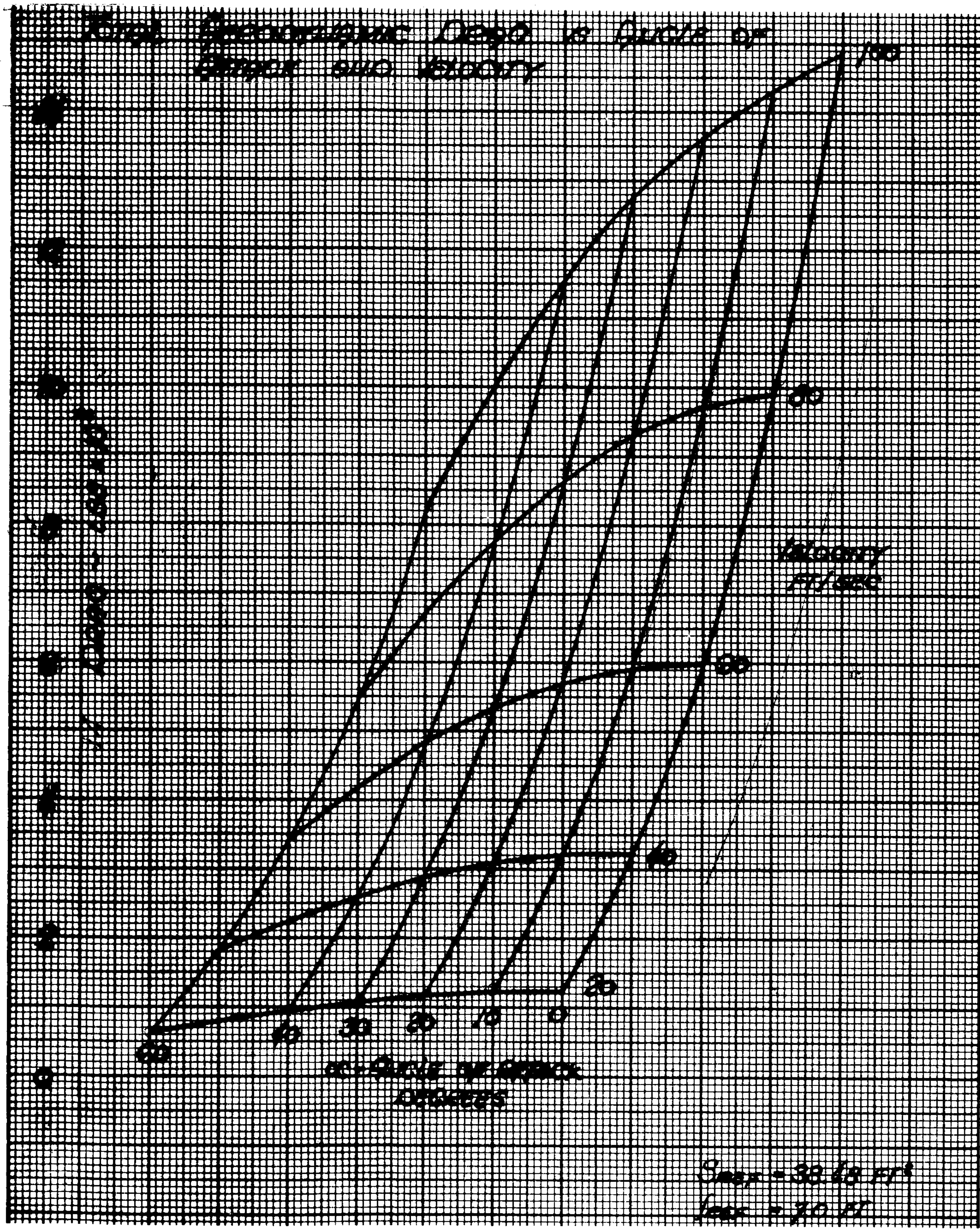


Figure IV-15. Total Aerodynamic Drag versus Angle of Attack and Velocity

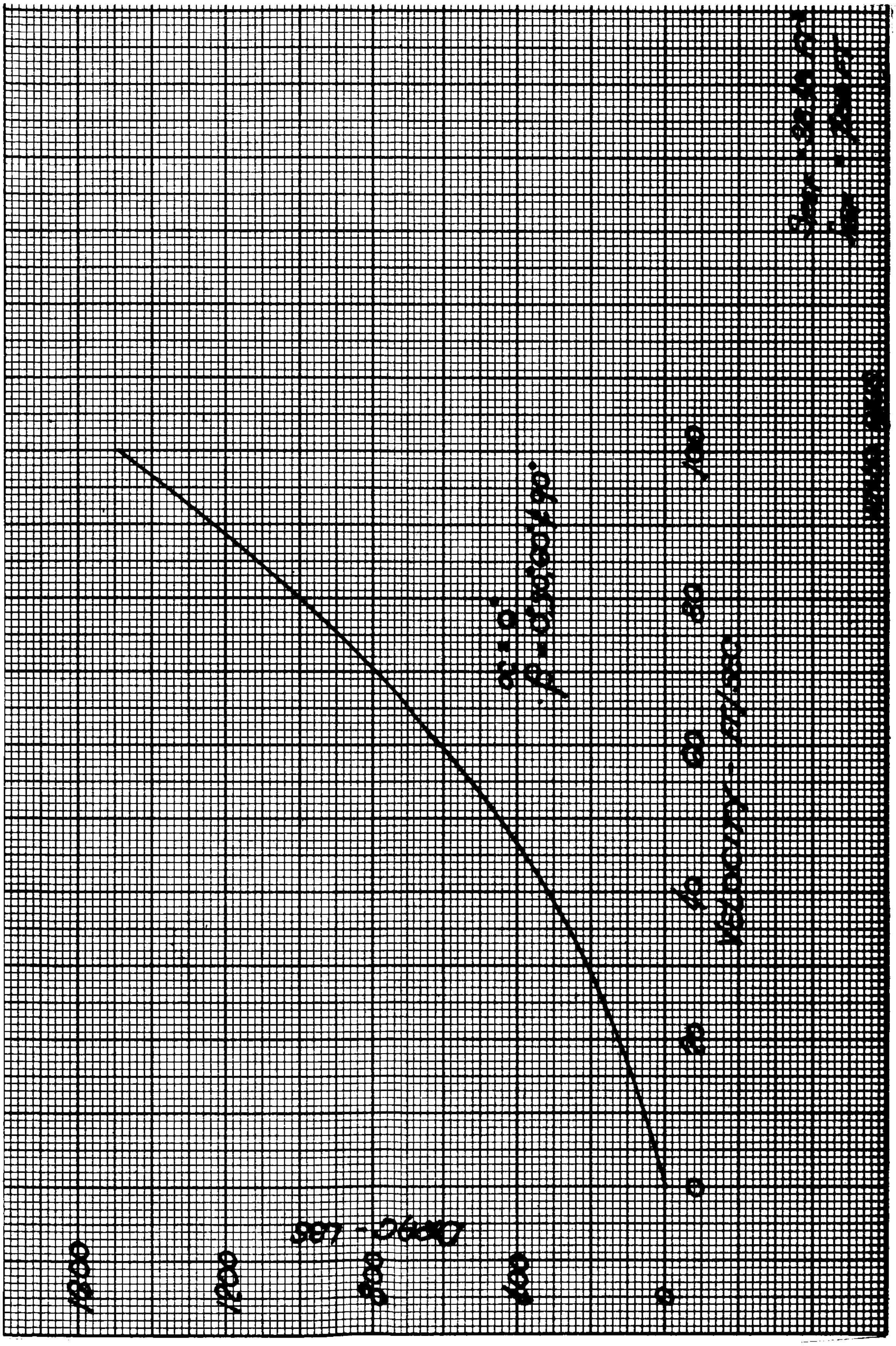


Figure IV-16. Effect of Sideslip Angle on Total Drag

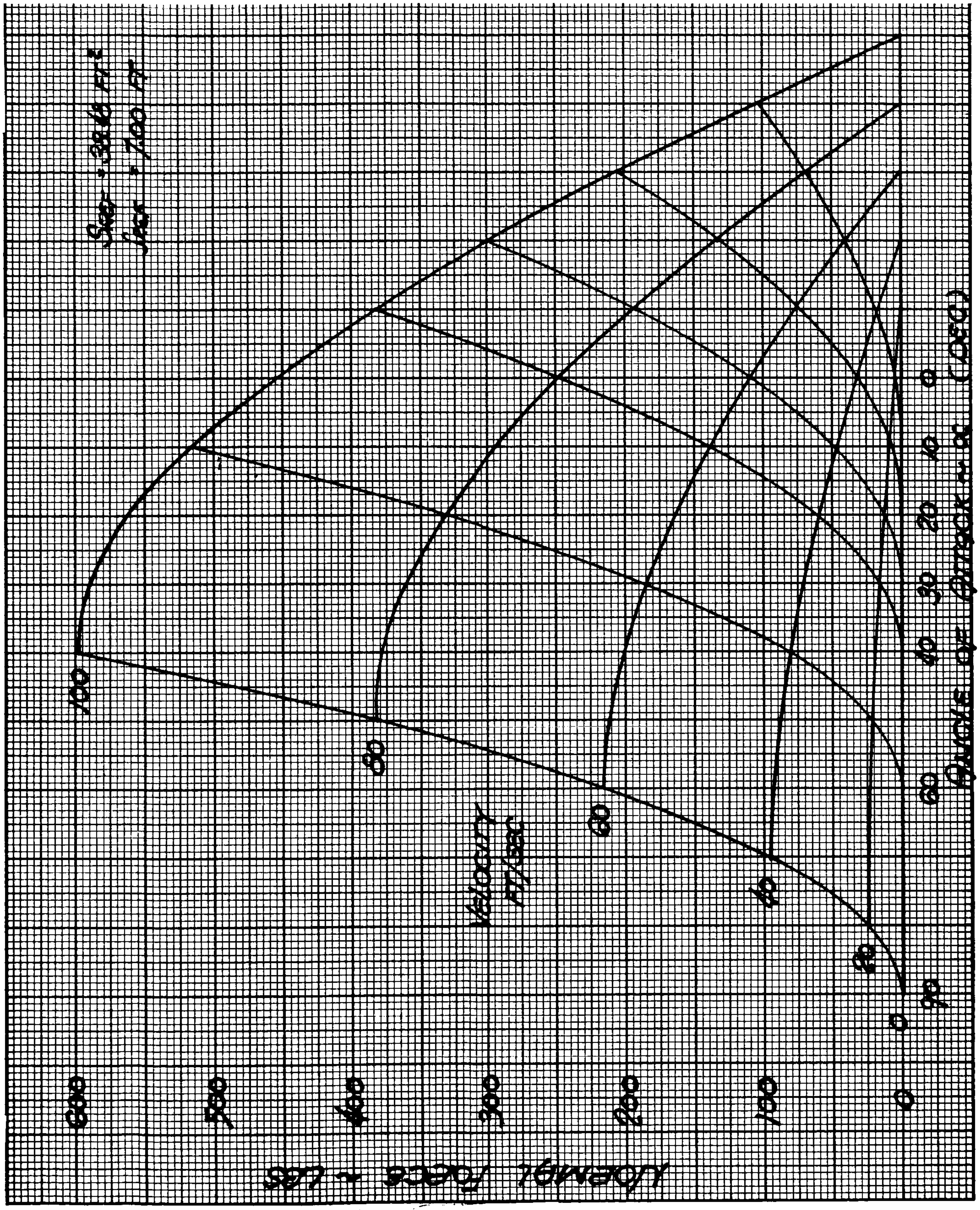


Figure IV-17. Normal Force versus Angle of Attack dependent, of course, upon total vehicle drag. The tilt angles required are shown in Figure V-7 as a function of vehicle velocity and are based upon the drag estimates just discussed. The limiting lateral velocity - from considerations of drag compensation - is the engine deflection angle. A tentative deflection limit of 12° has been established which would limit lateral velocity to approximately 70 ft/sec. The vehicle aerodynamic lift or normal force has a maximum value of approximately 150 lbs (.05-g's), at a vertical velocity of 50 ft/sec ($\alpha = 90^\circ$). This effect is compensated by the engine throttle command system.

The vehicle aerodynamic moments calculated for the present design configuration are shown in Figure IV-18. These moments are considerably greater than that desired for good simulation at the higher speeds. Analyses were, therefore, made of several design modifications to reduce these moments to tolerable levels. (Tolerable is construed to mean moments produced at the highest flight speed ($V \approx 70$ ft/sec) requiring only 10% to 15% of the single system attitude control available (220 to 330 ft/lb). Methods that were considered were: 1) greater inclination of the support legs with respect to the vertical, 2) reduced leg length, and 3) use of elliptic cross-sections for the struts. Of these methods, the greatest reduction in moment was achieved by the greater inclination of the support legs. The analysis showed that the three main struts on each leg contribute the largest moments. Increasing the inclination relative to the vertical has a powerful effect since their normal force decreases as well as the center of pressure location relative to the center of gravity. Figure IV-19 shows the reduction in moments versus velocity as the legs on the present configuration are moved out an additional 10° and 20° respectively. This reduces the moments substantially so that their effect on the vehicle is significantly reduced with a minimum configuration change, each of the four legs were reduced in length by an arbitrary 12 inches without altering their present position. These results indicated the per cent change in center of pressure to be quite small, approximately 2.5%, so that in order to obtain significant reduction in aerodynamic moments, a larger leg length reduction would have to be considered. (The effect of using elliptical cross-sections in the main support members was found to be detrimental. The cross-section size required for structural rigidity was such that the added drag and moments of the two legs in which the major axis was normal to the flow direction was greater than the reduction realized in the other two legs.) A comparison of these methods indicated that the first is much more effective in obtaining greater reduction in aerodynamic moments. The very large reduction in moments possible due to leg inclination, together with other reductions due to placing certain of the cross struts at more favorable inclination angles, indicate that aerodynamic moments can be reduced to tolerable levels. The theoretical moment predictions are considered

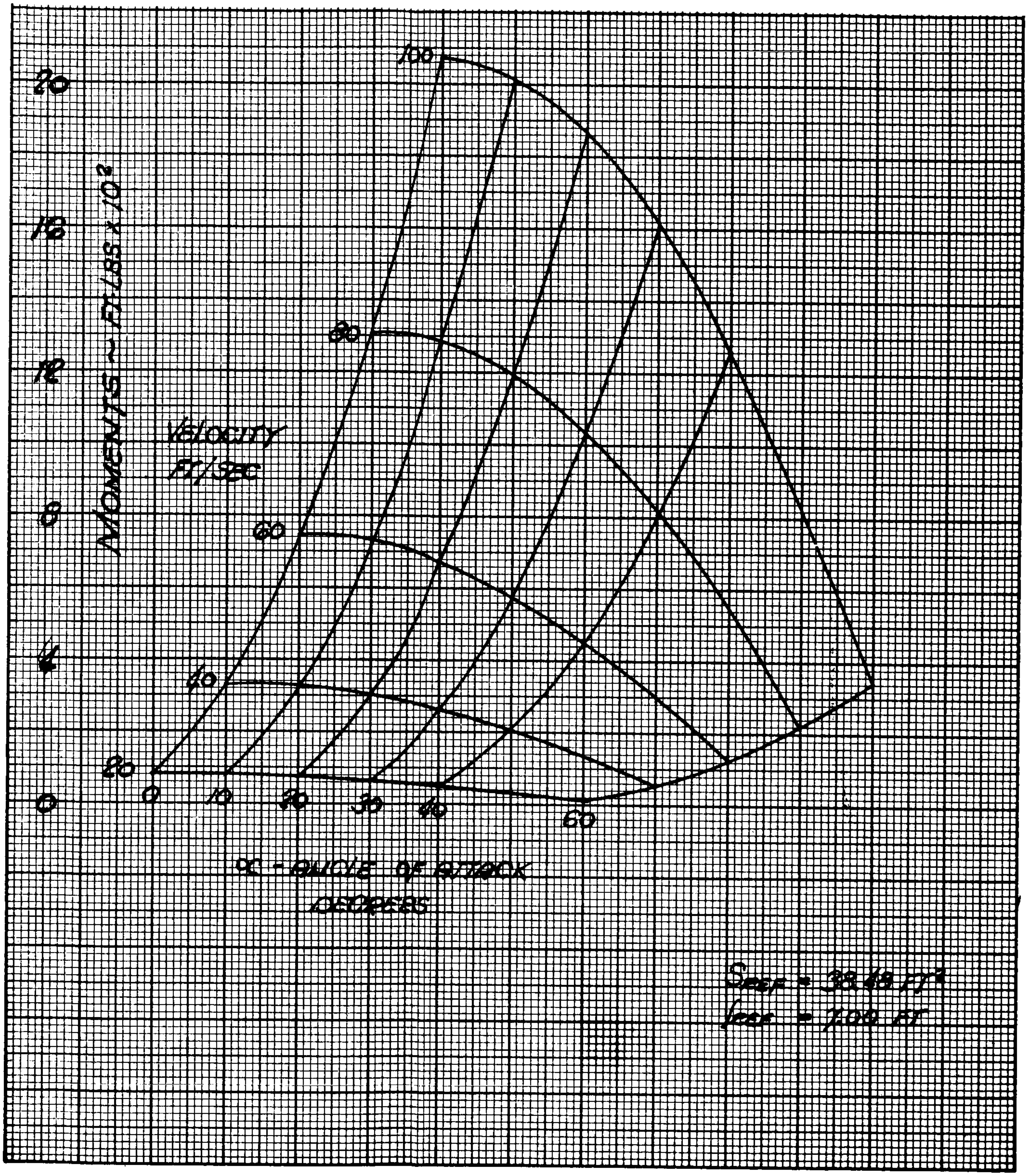


Figure IV-18. Total Aerodynamics Moment versus Angle of Attack and Velocity

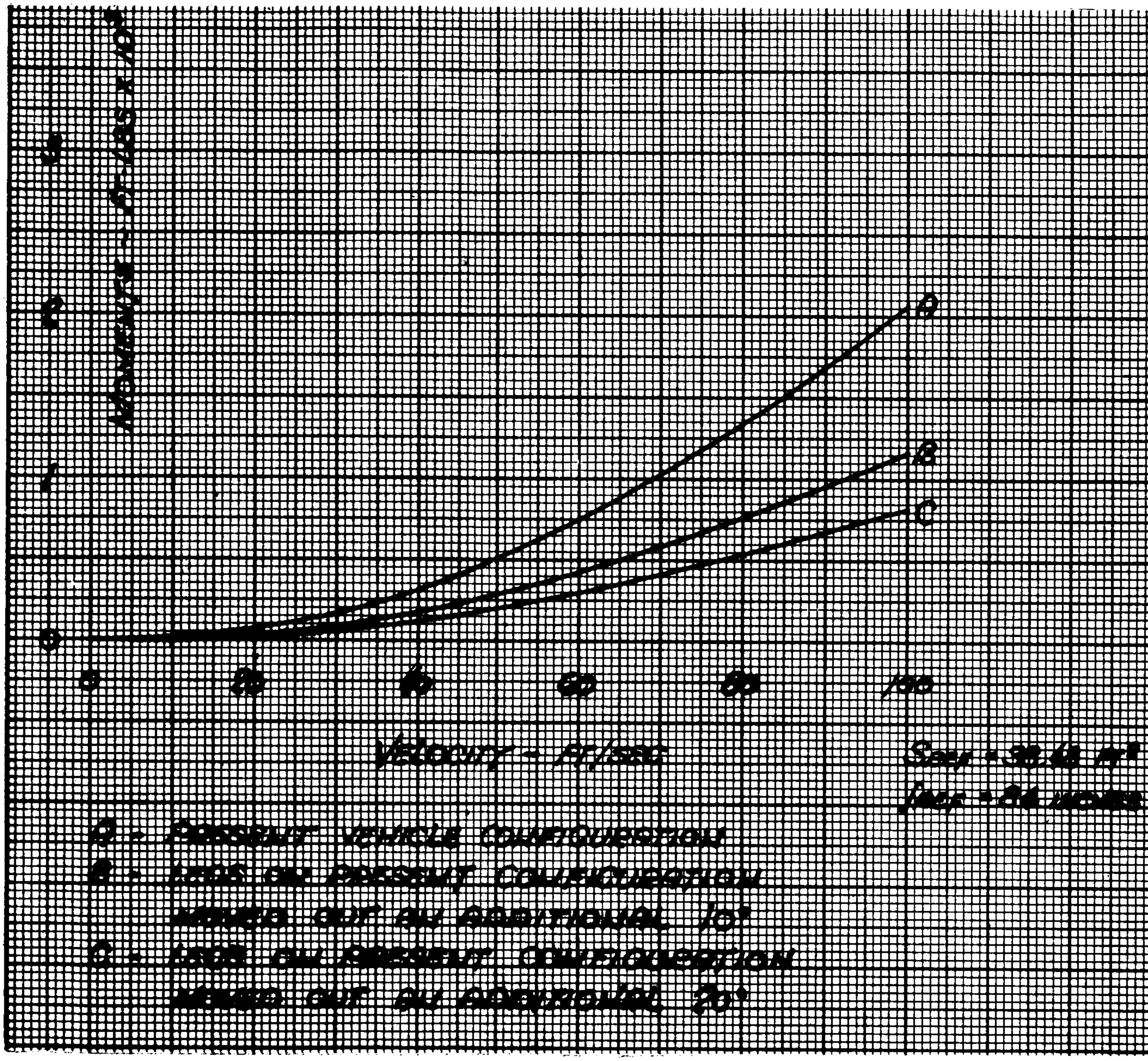


Figure IV-19. Effects of Various Leg Angles on Total Aerodynamic Moment somewhat conservative since interference drag will probably have its greatest effect in the upper vehicle regions such as to further reduce moments.

2. Engine Drag and Pitching Moments

The aerodynamic drag and pitching moments produced on the engine during horizontal translational flight result from two sources. One is the drag and pitching moment produced by the momentum change due to turning the air into the engine intake, and the other is that due to the external drag of the engine and its components. Of these, the drag and moments due to momentum change are much greater. The engine configuration that was selected for the lunar landing simulator has two inlets - turbojet and turbofan - which permit locating the gimbal axis so that the moments due to momentum drag of each inlet offset each other to a large degree. The ideal situation would, of course, be a gimbal axis location which not only balances all aerodynamic moments to zero but also all inertial moments, i.e., on the engine center of gravity.

An important factor in determining the magnitude of moment due to momentum drag is the shape of the inlet. A large radius inlet will develop larger aerodynamic moments because of the additional pure couple that is produced by the suction pressures acting on the windward lip and the stagnation pressures acting on the leeward lip. Estimates were made of the drag and pitching moments of the selected engine to inlet momentum change based upon wind tunnel test data (NASA TN D-995) and other Bell Aerosystems Company tests of powered ducted fan model configurations, in which the engine axis was normal to the flow direction. These data covered a range of thrust coefficients and inlet shapes generally representative of the present engine. These data were used to correlate the effective center of pressure location (above the inlet) as a function of thrust coefficient and inlet shape. The moment produced due to translational velocities (normal to engine axis) is then equal to the momentum drag, $\dot{m} V_\infty$, (mass flow times free stream velocity) multiplied by the vertical arm, $\bar{Z}$, from center of pressure to gimbal point. Moment $= \dot{m} V_\infty \bar{Z}$, the additional aerodynamic drag and moment due to external flow were estimated by determining the cross flow drag of the two major engine segments treated as effective cylinders. A conservative cross flow drag coefficient of 1.0 was used to account for equipment appendages (even though Reynolds Numbers are above critical at moderate translational speeds). The results of these moment and drag calculations are presented (for two gimbal axis locations) in Figure IV-20. The variation of aerodynamic moments with velocity is shown to flatten out and even reduce at the higher speeds. This is due to the fact that the effective center of pressure moves closer to the engine lip as free stream dynamic pressures increase.

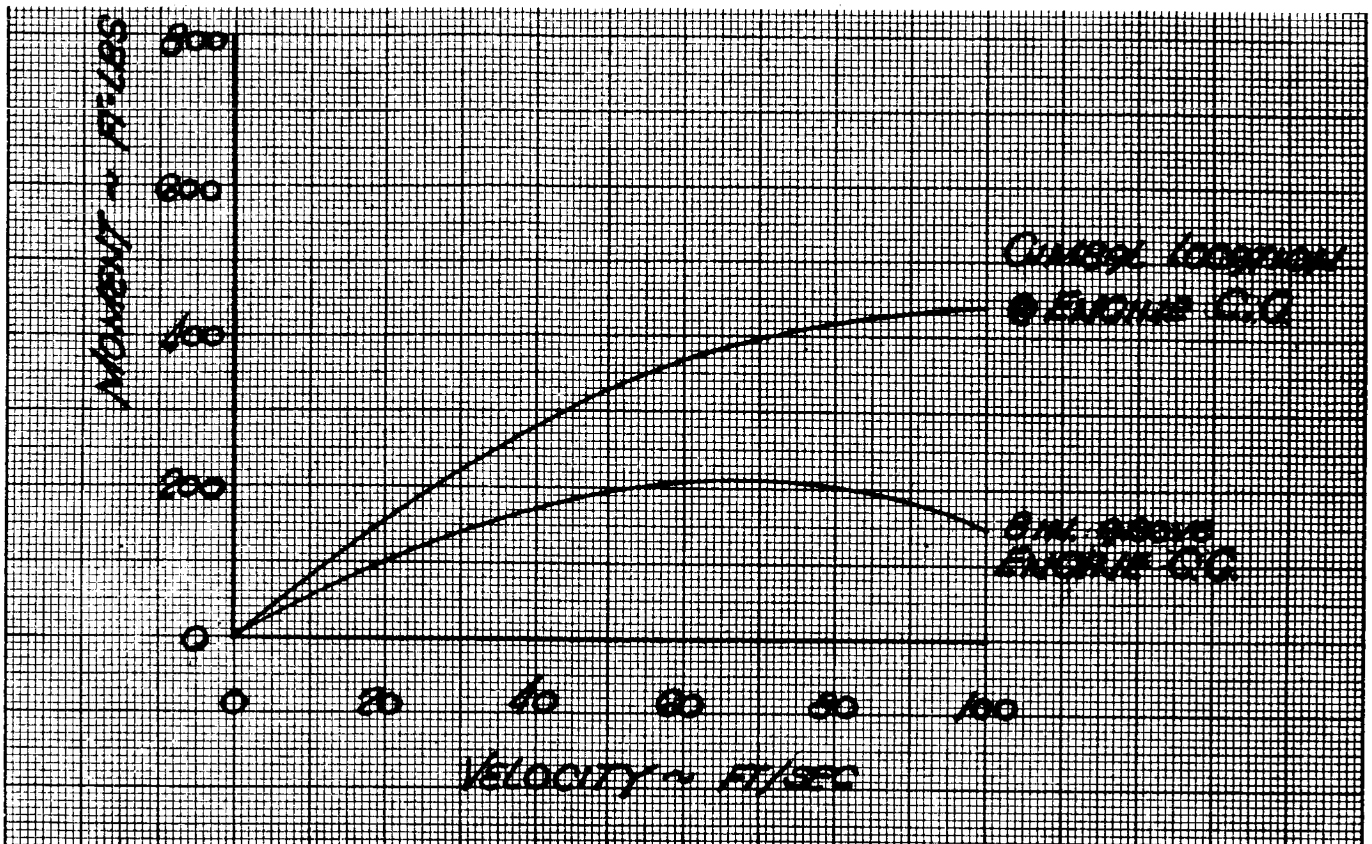
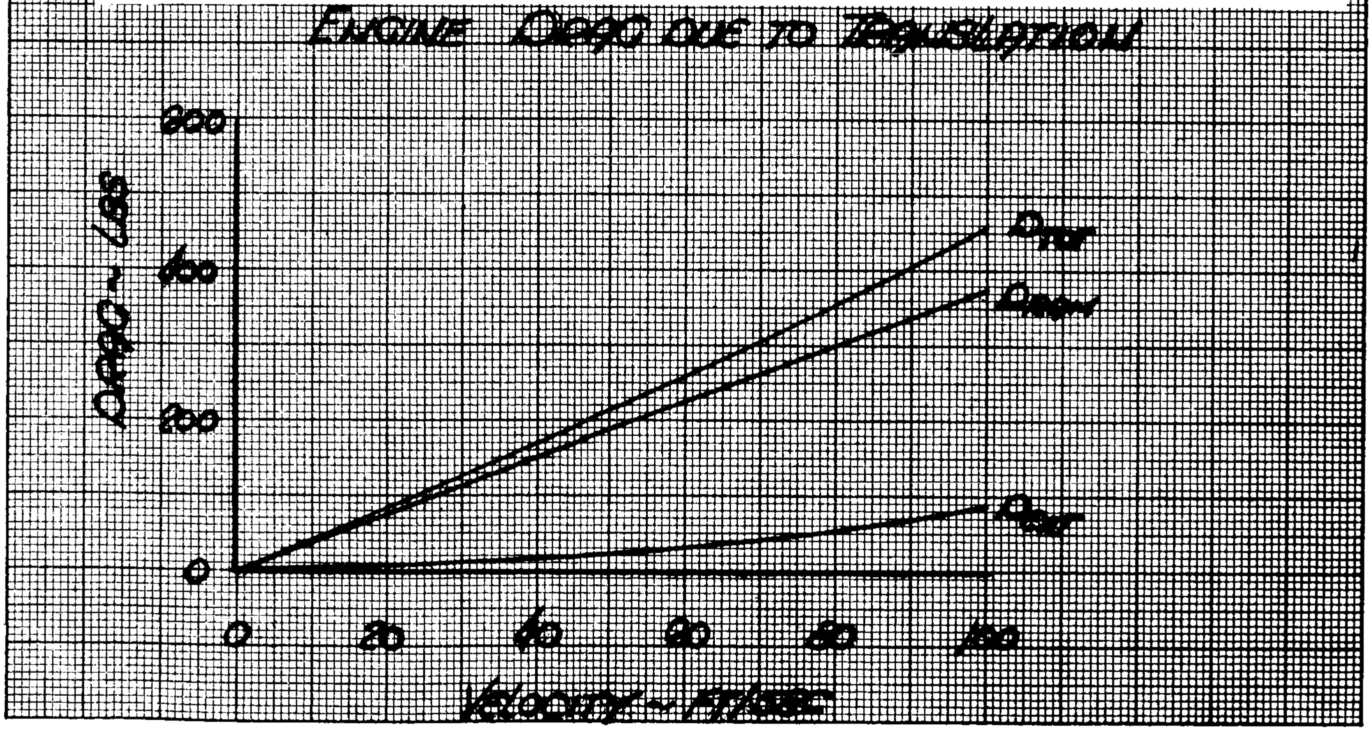


Figure IV-20. Engine Aerodynamic Pitching Moments Due to Translation



The aerodynamic moments for a gimbal axis located at the engine center of gravity are fairly large (400 ft/lb at V = 70 fps) and positive (nose up). It was therefore considered advisable to select a gimbal axis location above the engine center of gravity such as to reduce aerodynamic moments and thus optimize the sum of aerodynamic and inertial moments. Figure IV-21 shows this variation of aerodynamic and inertial moments with gimbal axis location. The inertial moments were based upon a .2-g maximum horizontal acceleration condition which would occur with the vehicle tilted 45^{o} at full thrust (T = 2/6 W). The 8-inch location is shown to be near optimum and would produce a maximum total moment (aerodynamic and inertial) of approximately 300 ft/lbs, which is compatible with the engine stabilization control power.

It may be noted that the fuel tank is placed above the gimbal axis so that with gross fuel, the engine plus fuel center of gravity is on the gimbal axis. Thus, the maximum inertial moment would only occur for maximum translational conditions near a fuel empty condition. An added advantage appears to occur from placing the gimbal axis above the engine center of gravity. That is, in the event of an engine stabilization failure, the engine system will be statically stable about a position aligned with the capsule vertical centerline. Further simulation studies will determine whether this is a safely controllable condition for this type of emergency.

Additional engine moments

Other sources of moments are:

a. Center of Gravity Offset
b. Gyroscopic Torque
c. Jet Rotor Countertorque

The engine center of gravity is estimated to be held within .2 inches of the thrust axis by suitable locating of auxiliary equipment. Assuming a maximum vertical acceleration of 1.33-g's (1.0-g from main engine and 2/6-g from capsule thrusters), the engine moment produced would be 17 ft-lbs. Moments due to gyroscopic torques have been calculated and are found to be very small. For an engine rotation rate of 10^{o}/sec, the gyroscopic torque amounts only to 4 ft-lbs. No estimates have yet been attempted or information received to determine the jet rotor countertorque, i.e., torque about the engine centerline, however, its effect is expected to be small. A tabular summary of engine moments for most critical operating conditions is presented below.

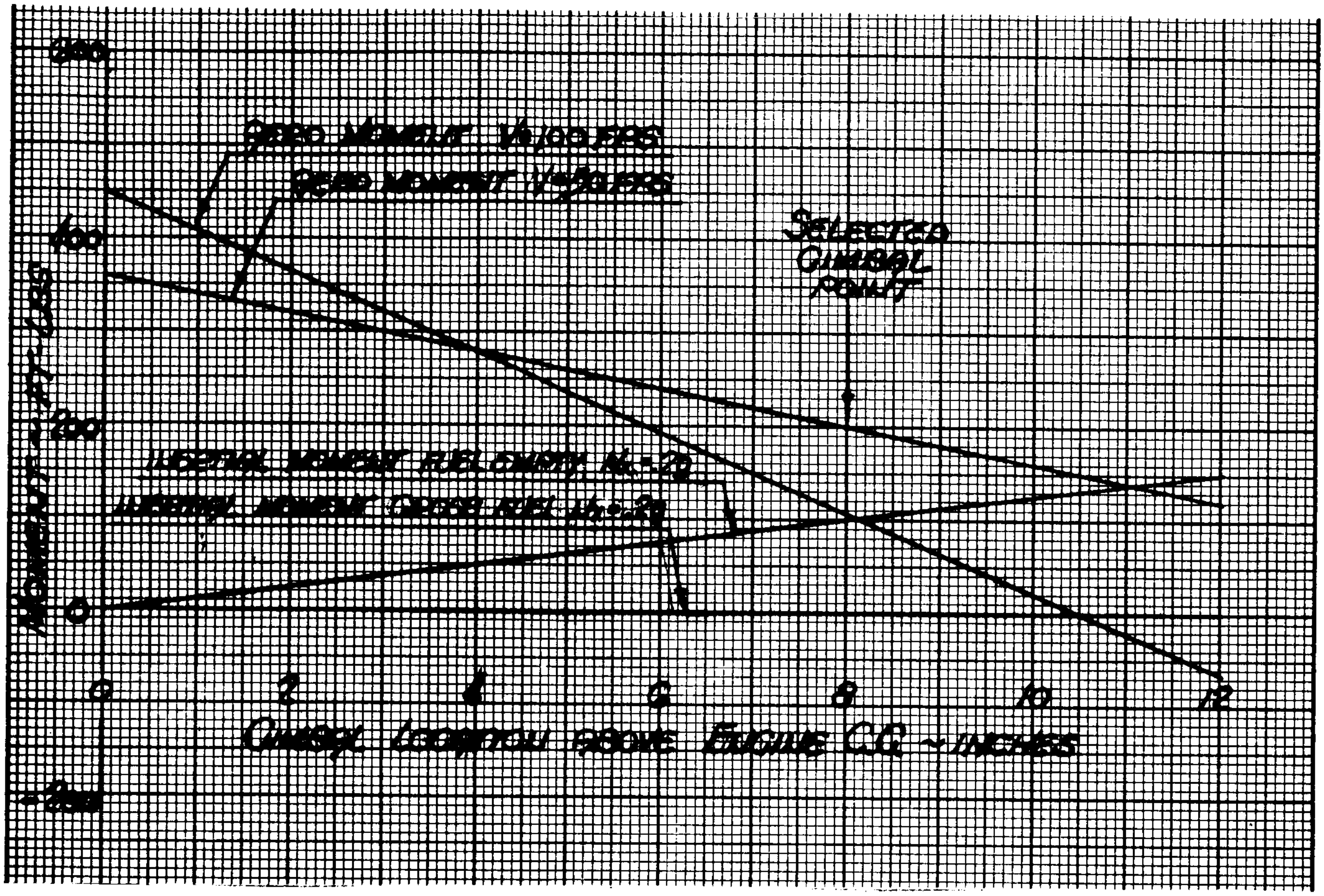


Figure IV-21. Variation of Engine Aerodynamics and Inertial Pitching Moments with Gimbal Location

Moment Source	Operating Condition	Moments
Aerodynamics	Horizontal Velocity = 70 fps	210 ft-lbs
Lateral Acceleration	N_x = .2-g	100 ft-lbs
Vertical Acceleration	N_z = 1.33-g	17 ft-lbs
Gyroscopic	$\dot{\theta}$ engine = 10°/sec	4 ft-lbs

Although an unlikely combination of circumstances would be required to produce all moments additive in one direction, the total moment produced would be 331 ft-lbs. The stabilization control moment available from the bleed control system is 480 ft-lbs. It may be noted here that the engine aerodynamic moment acts in such a direction, (nose up) to reduce vehicle speed should a speed condition develop in which the moments exceed the stabilization system control power, i.e., the system is speed stable.

E. ANALYSIS OF CONTROL LOOPS

1. Vehicle Stabilization and Variable Stability Control System

Figure IV-22 is a block diagram of the vehicle pitch attitude control system. The analysis also applies to the roll and yaw attitude control systems.

The definition of the nomenclature used is as follows:

Symbol		
K	Control stick potentiometer sensitivity	volts/rad
K_θ	Attitude gyro sensitivity	volts/rad
τ_g	Rate gyro sensitivity	volts/rad/sec
I_y	Moment of inertia of the vehicle	slug ft^2
K_t	Torque sensitivity of the hydrogen peroxide jets and valve	lb ft/volt
τ_v	Time lag of the hydrogen peroxide valve and combustion chamber	second
T_{max}	Maximum torque output of the jets	lb ft
V_E	Error voltage	

Because the error voltage (V_E) commands a proportional pulse duration of the torque, the torque sensitivity (K_t) is a smoothed term which approaches the linearized approximation for time periods which are several times the value of the pulse width.

Aerodynamic, static, and dynamic unbalance moments and forces are considered to be effectively compensated for in this system.

When operating with zero attitude and rate gyro feedback, the pilot commands vehicle angular accelerations directly. The sensitivity of vehicle angular acceleration to control stick motion can be varied from zero to a maximum value as determined from the resolution of the control stick potentiometer, and the maximum torque capability of the reaction jets. By using an induction potentiometer on the control stick the maximum acceleration sensitivity of the system will only be limited by the pilot's threshold in moving the control stick or in visually resolving an error signal.

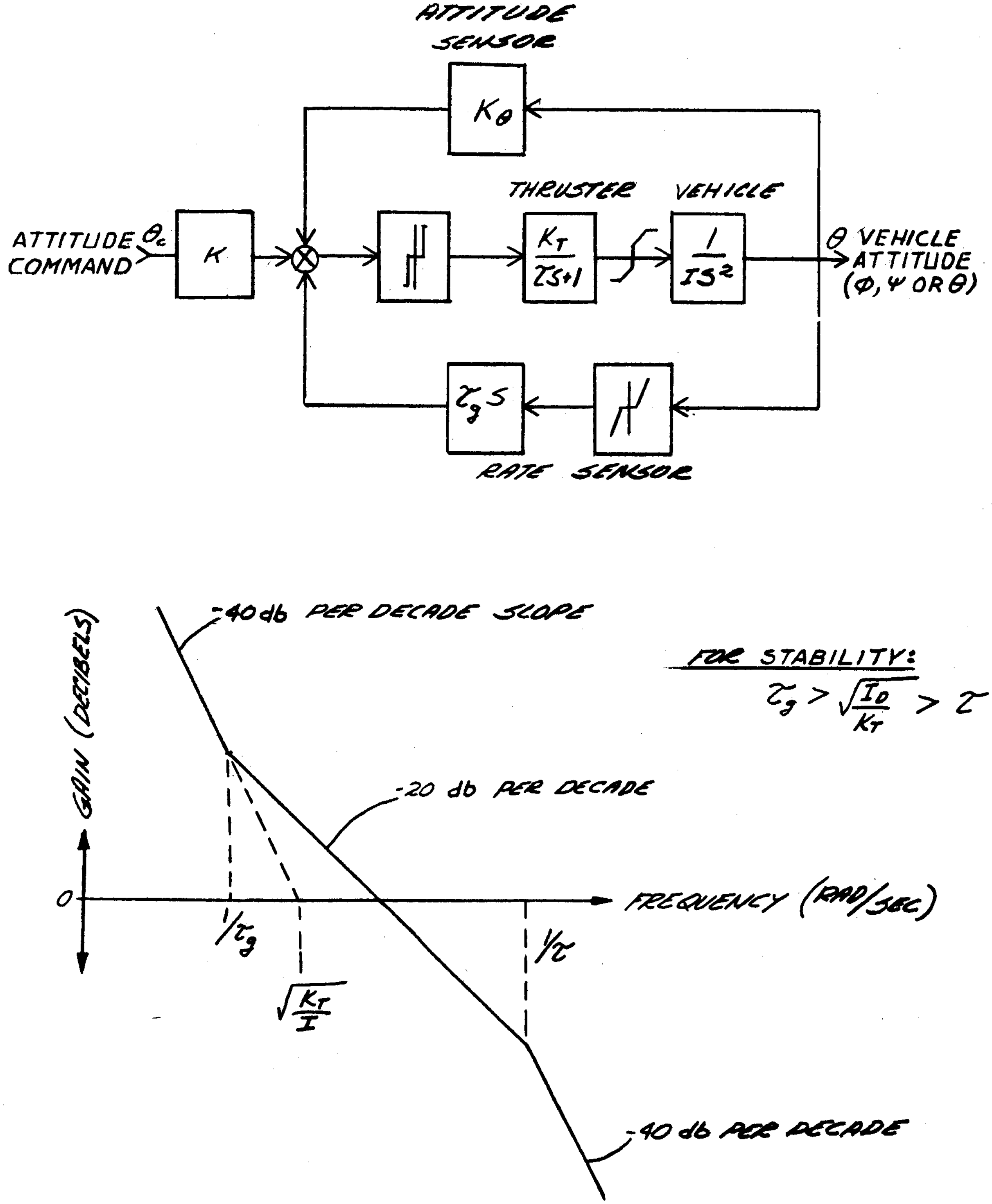


Figure IV-22. Pitch Control System

An examination of the equations for this configuration shows that the pilot must provide the damping and attitude signals required for stability. The inertia of the pilot's hand and control stick in conjunction with his response time will place an upper limit on the maximum acceleration sensitivity that he can effectively handle. Conventional displacement sticks and force sticks, as well as variable feel devices, can be used to evaluate the extent of this limitation.

The pilot's capability to provide damping to the system can be enhanced by supplying him with visual information (indicator) on attitude rate. The window of the vehicle and the turn and bank indicator supplies the visual attitude information needed by the pilot.

Artificial damping can be provided to the system by increasing the rate gyro feedback (τ_g). When this feedback is negative, positive damping is added to the system. However, if the feedback is made positive, the pilot will have to provide even greater damping to the system.

With rate gyro feedback the control system is changed from an acceleration command to a velocity control system. The closed loop transfer function for the control system is:

$$\frac{\theta}{\theta_c} = \frac{K/\tau_g}{s\left(\frac{I\tau_v}{K_t\tau_g}s^2 + \frac{I}{K_t\tau_g}s + 1\right)}$$

A large response bandwidth for this system requires a high loop gain ($K_t\tau_g$); however, as can be seen from the above equation, the inertia of the vehicle (I) and valve time constant (τ_v) limit the maximum loop gain if we are to maintain the resonant peak reasonably damped, i.e. for a damping ratio (ζ) > 0.5 of critical the rate loop gain $(K_t\tau_g) < I/\tau_v$. For this condition the angular rate output of the system is attenuated at frequencies above $1/\tau_v$ radians per second.

Introducing attitude feedback into the system allows the pilot to command attitude directly with his control stick, i.e. each position of the control stick corresponds to a specific tilt angle of the vehicle. By making the attitude feedback signal positive, a statically unstable vehicle can be simulated, i.e. (center of pressure ahead of center of gravity).

With negative attitude feedback, the previous velocity command system is changed to an attitude or position control system. The closed loop transfer function for the attitude control system is:

$$\frac{\theta}{\theta_c} = \frac{1}{\frac{I \tau_v}{K_t} s^3 + \frac{I}{K_t} s^2 + \tau_g s+1}$$

Examining the characteristic equation for this system, the 180 degree phase shift frequency ($W_{180°}$) occur at $\sqrt{\frac{K_t}{I}}$ radians per second, and the 270° phase shift frequency occurs at $\sqrt{\frac{\tau_g}{\tau_v}}\, W_{180°}$. For a smooth operating system it is desirable that the 180 degree phase shift frequency be lower than the 270 degree frequency or the rate feedback time constant be larger than the valve time lag ($\tau_g > \tau_v$).

If in this system the torque sensitivity (K_t) is made sufficiently large, the above equation reduces to a simple first order lag with a break frequency of $1/\tau_g$ radians per second.

Associated with a specified thrust level and configuration of a reaction jet, there is a minimum controllable impulse below which it is not practicable to try to control, see Figure IV-23. For the eighty pound hydrogen peroxide thrusters the minimum controllable impulse is about 1.16 pound second. This corresponds to a threshold angular rate of 0.55 degree per second about the pitch or roll axis.

Considering a ±1.8 degree allowable deadband and the threshold angular rate, the period of the limit cycle would be 13 seconds. The period of the limit cycle can be made longer by using vernier thrusters or by increasing the allowable deadband. Angular rate feedback has the effect of firing the jets before the attitude deadband limit is reached and thereby preventing overshoots. To be effective, however, the rate sensor threshold should be lower than the reaction jet threshold. High rate feedback in the system has the effect of commanding the minimum controllable impulse per cycle, and of preventing the limit cycle from diverging.

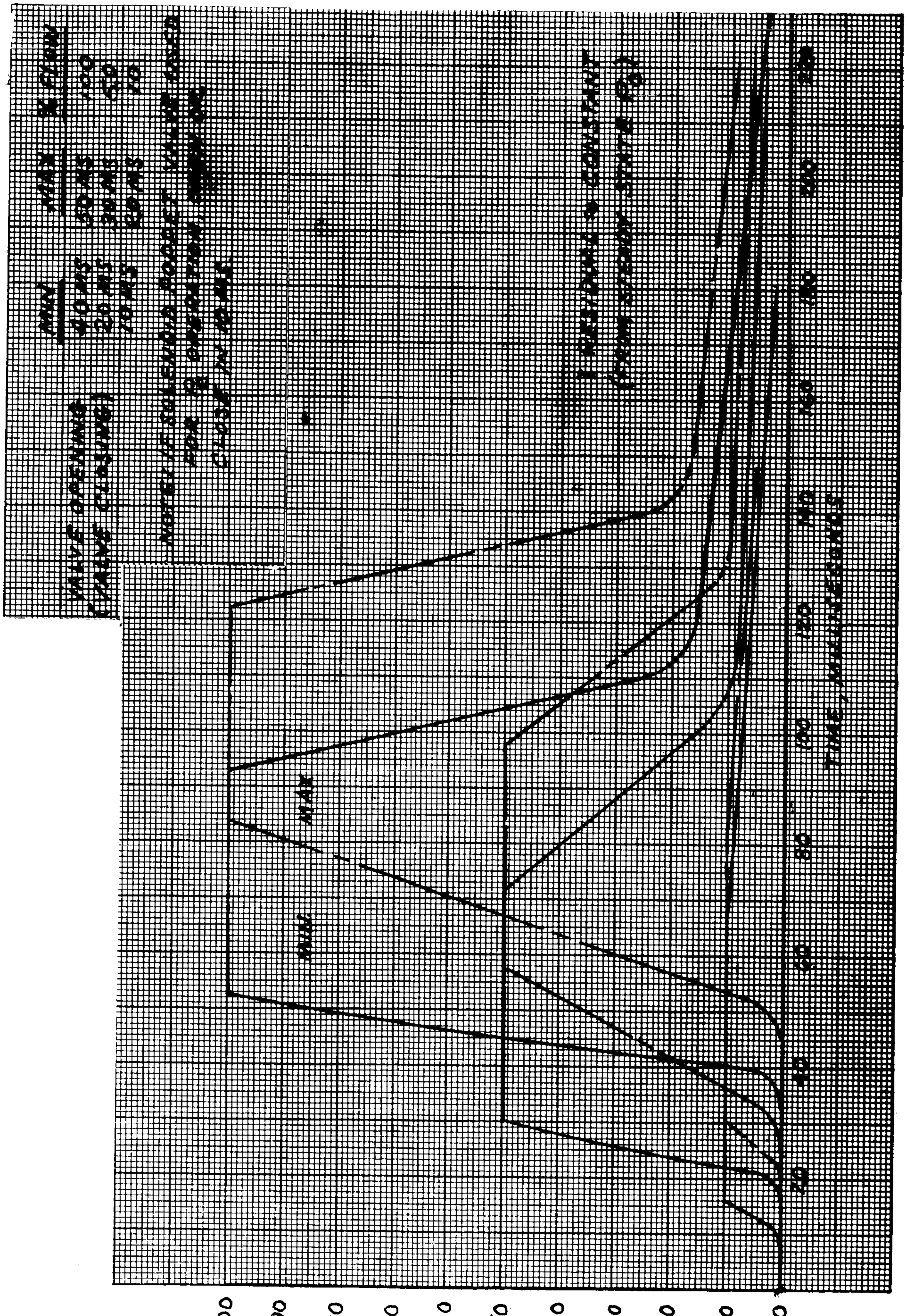


Figure IV-23. Impulse Bit Envelope, Motor Driven Ball Valve

2. Jet Engine Throttle Control System

The system rate, stability and accuracy considerations may be discussed briefly if reference is made to the block diagram of Figure IV-24. In the command mode the basic closed loop equation is:

$$\frac{\theta}{\theta_c} = \frac{\frac{1}{K_{fb}\, K_{gl}}}{\frac{\tau_m}{K_a K_m K_{fb} K_{gl}}\, S^2 + \frac{S}{K_a K_m K_{fb} K_g}\,(1 + K_a K_m K_{\dot{\theta}}) + 1}$$

To satisfy the 0.26 second system response time requirement, the closed loop break frequency w_n should be equal to or greater than:

$$\omega_n \geq \frac{1}{0.26} \geq \sqrt{K_a \left(\frac{K_m}{\tau_m}\right) K_{fb} K_{gl}}$$

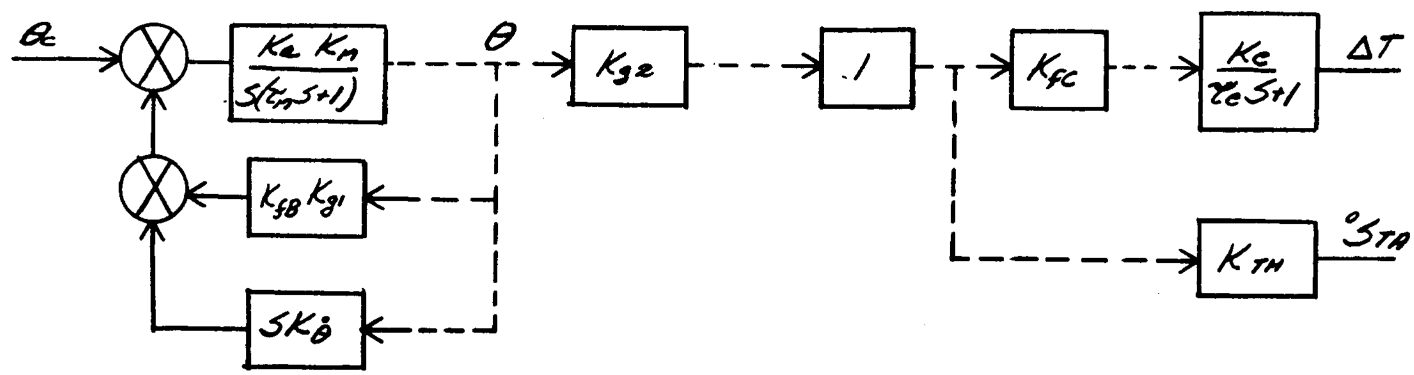
The damping gain $K_{\dot{\theta}}$ is adjusted to establish a damping coefficient of between 0.4 and 0.7.

$$\zeta = \frac{1 + K_a K_m K_{\dot{\theta}}}{\sqrt{K_a K_m K_{fb} K_{gl} \tau_m}}$$

The constants given in the above equations are straight forward and are defined in Figure IV-24. However, it should be pointed out that the friction load in the system will have to be determined for the proper determination of K_m. In addition, the inertia loads should be reflected back to the servo motor shaft to determine if their influence on τ_m. The engine time constant τ_e should not exceed 16 seconds for a thrust change of 1500 pounds to eliminate the need for a compensating network.

3. Engine Bleed Jet Control System

A control system for the vertical attitude of the jet engine was designed and simulated on the analogue computer. The analogue simulation circuit for this control system appears in Figure IV-25. The portion within the dotted lines pertains to the physical system, i.e. the engine. A is the moment on Inertia about X which equals 34.2 slug ft^2, and B is the moment of Inertia about Y which equals 41.5 slug ft^2. No time scaling was performed since the reaction times of the system were compatible with recording equipment available.



K_a · AMPLIFIER GAIN (VOLTS/VOLT)
K_m · MOTOR GAIN (RAD/SEC/VOLT)
τ_m · MOTOR TIME CONSTANT (SEC)
K_{fB} : POSITION FEEDBACK (VOLTS/RAD)
K_{g1} : REDUCTION GEAR RATIO (RAD/RAD)
$K_{\dot{\theta}}$ = RATE FEEDBACK (VOLTS/RAD/SEC)
K_{g2} = REDUCTION GEAR RATIO (RAD/RAD)
K_{fc} = FUEL CONTROL LINKAGE RATIO (RAD/RAD)
K_e = ENGINE GAIN (POUNDS THRUST PER RADIN OF FUEL CONTROL)
τ_e = ENGINE TIME CONSTANT (SEC)
K_{TH} = THROTTLE GAIN (RAD/RAD)

Figure IV-24. Jet Throttle Control Block Diagram

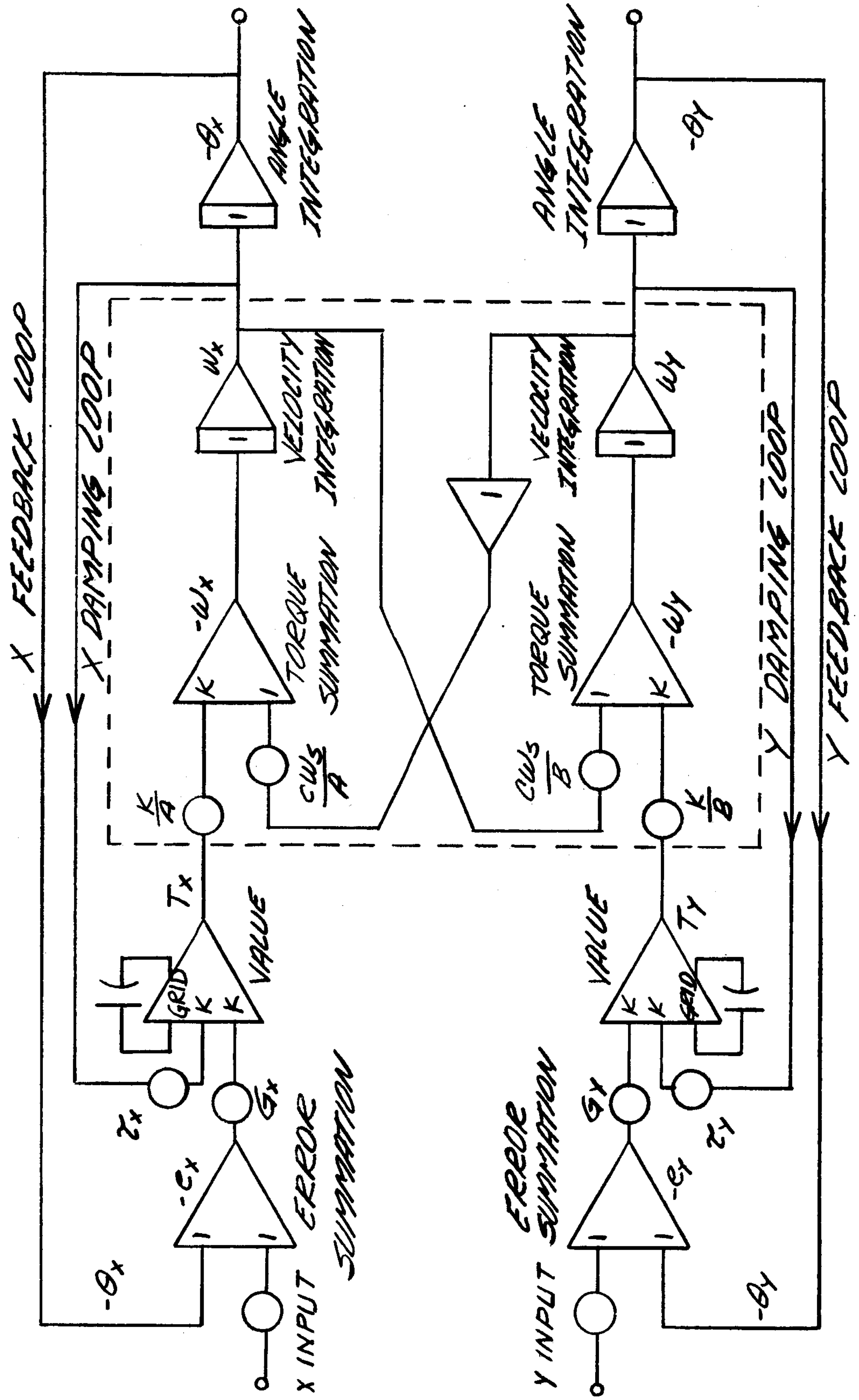


Figure IV-25. Jet Stabilization System Schematic

Response to Air Drag and Torque

Air drag disturbances will be transduced into the form of angular attitude commands for the engine. To determine the response to drag disturbances, step inputs of ten volts were interjected into the inputs, a correspondence of one volt per degree was assumed. The inputs and the outputs were recorded and appear in Figure IV-26 and IV-27. Considerable experimentation was performed to establish the gain and damping that yielded the fastest response time. Increasing the gain of the system would increase the tendency to oscillate, and increasing the damping sufficient to insure stability increases the response time. The result is a slower system. Decreasing the gain results in a sluggish system regardless of how slight a damping is allowed. The gain of 2000 ft lbs per degree and the damping of 600 ft lbs per degree per second were found optimum. The curves were recorded at ten units per second so that the response time is about .45 sec.

Also included in the Figures IV-26 and IV-27 are the integrals of the orthogonal errors. These integrals correspond directly to the orthogonal velocity that will be imparted to the vehicle by the orthogonal error. Since the error integrals equal .35 degree seconds and one degree of error will yield 1/60th of a 'g' acceleration, computation will verify that this velocity will be about 1.5 inches per second. This is rather reasonable for a ten degree step input. It was found that this velocity was a decreasing function of gain and independent of damping.

To test the response to torque disturbances, a torque input was put directly into the torque summation amplifiers in the form of a step voltage corresponding to 100 ft lbs of torque. The resultant errors appear in Figures IV-28 and IV-29. The maximum error is .07 degrees, or about 4 minutes of arc. This is a decreasing function of gain, and independent of damping.

For the actual system a check was made to determine the feasibility of the gain value previously established as 2000 ft lbs per degree. A steady state torque of 300 ft lbs was assumed which leaves 180 ft lbs for dynamic stabilization, since a maximum of 480 ft lbs of torque is available from the air bleed system. It was observed for a step input of ten degrees the limiting torque of 480 ft lbs was not reached. Thus, this figure of gain is indeed practical.

Lack of Axial Symmetry and Adequacy of 2 Gimbal Mounting

Equations used in the simulation described above assume that the engine has axial symmetry. Since this is not the case, a

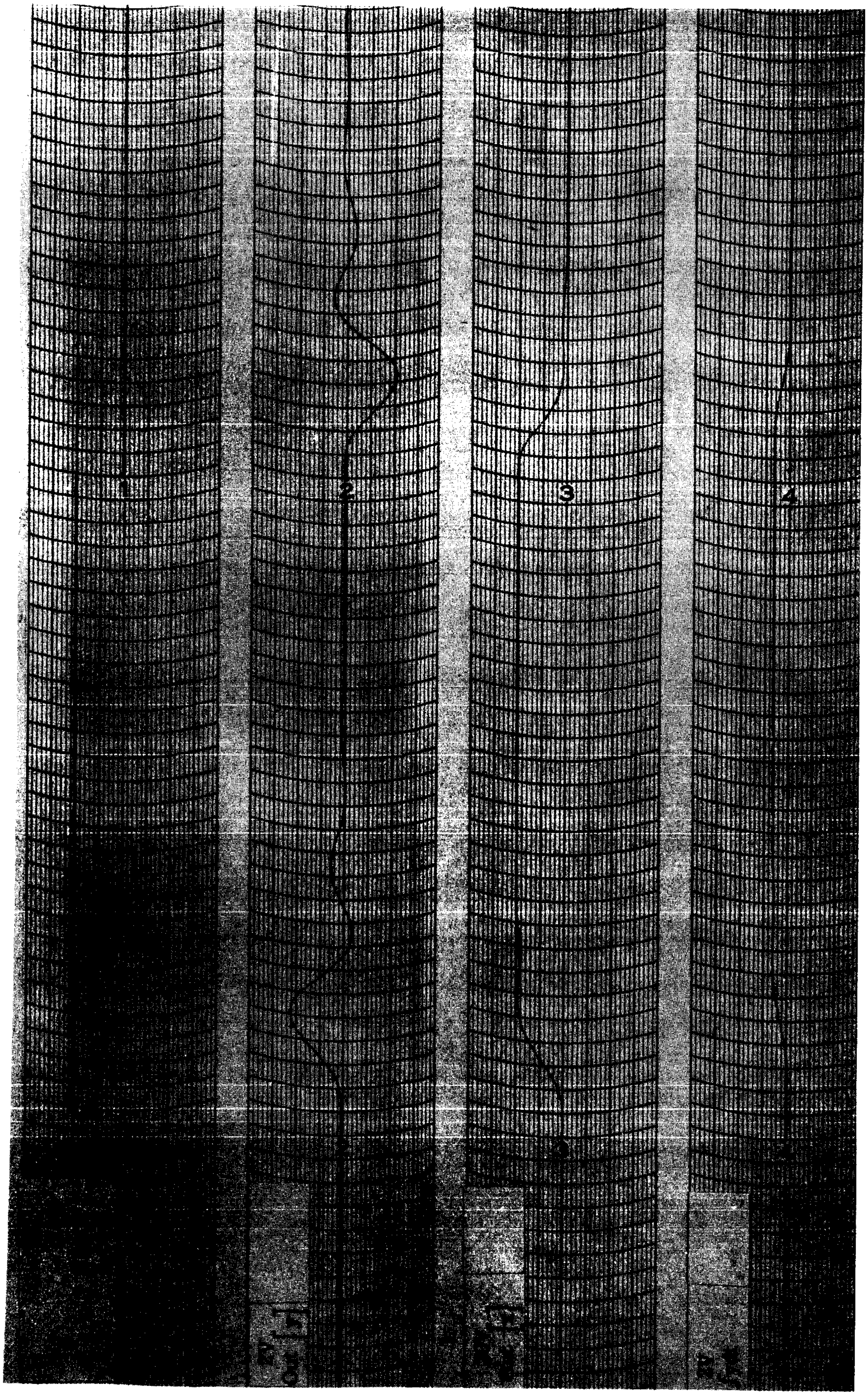


Figure IV-26. Response to 10° Step Input x - Gain Loop = 2K ft lb/deg Rate Damping = 600 ft lb/deg/sec

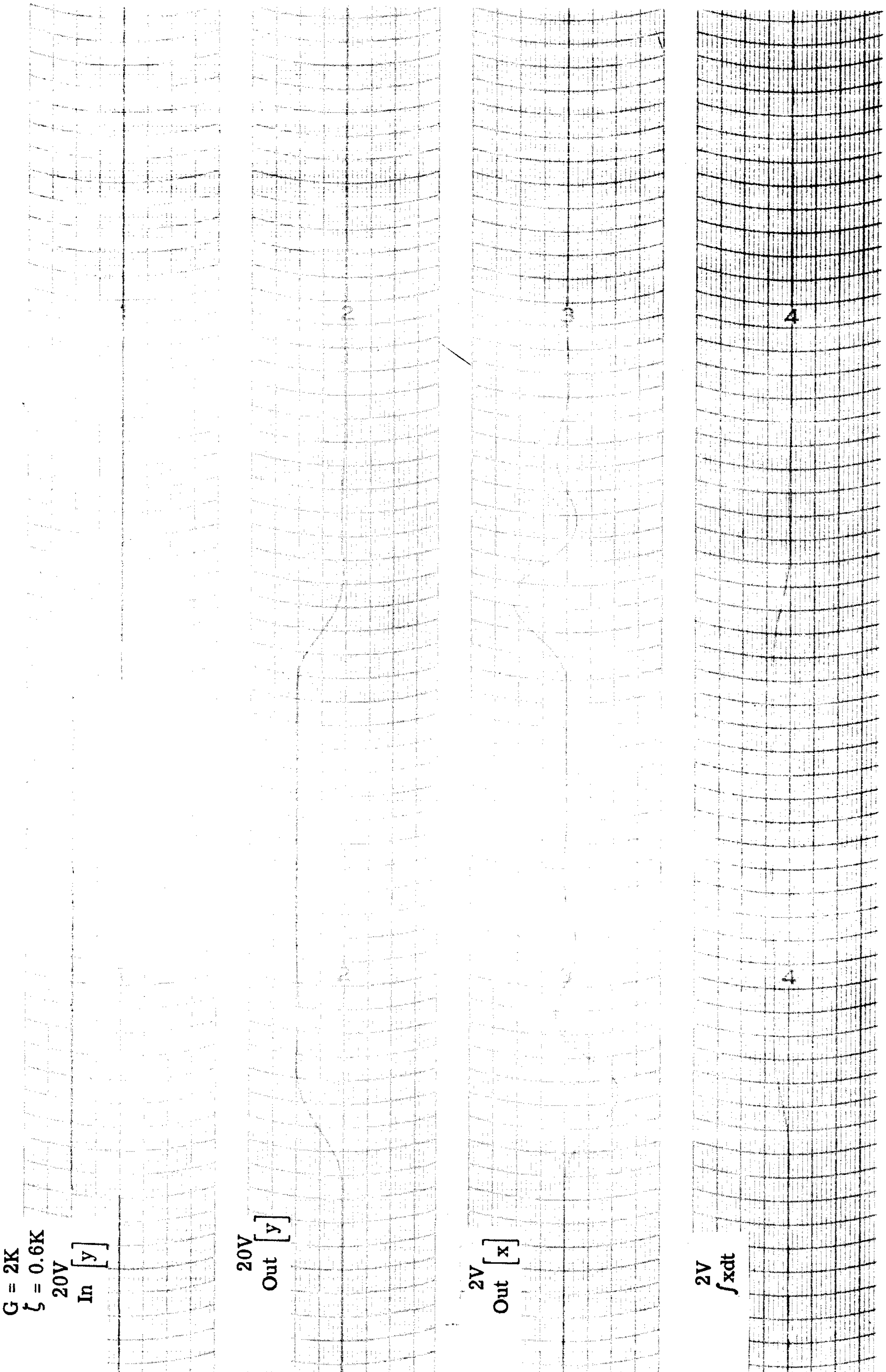


Figure IV-27. Response to 10° Step y Input — Gain Loop = 2K ft lb/deg Rate Damping = 600 ft lb/deg/sec

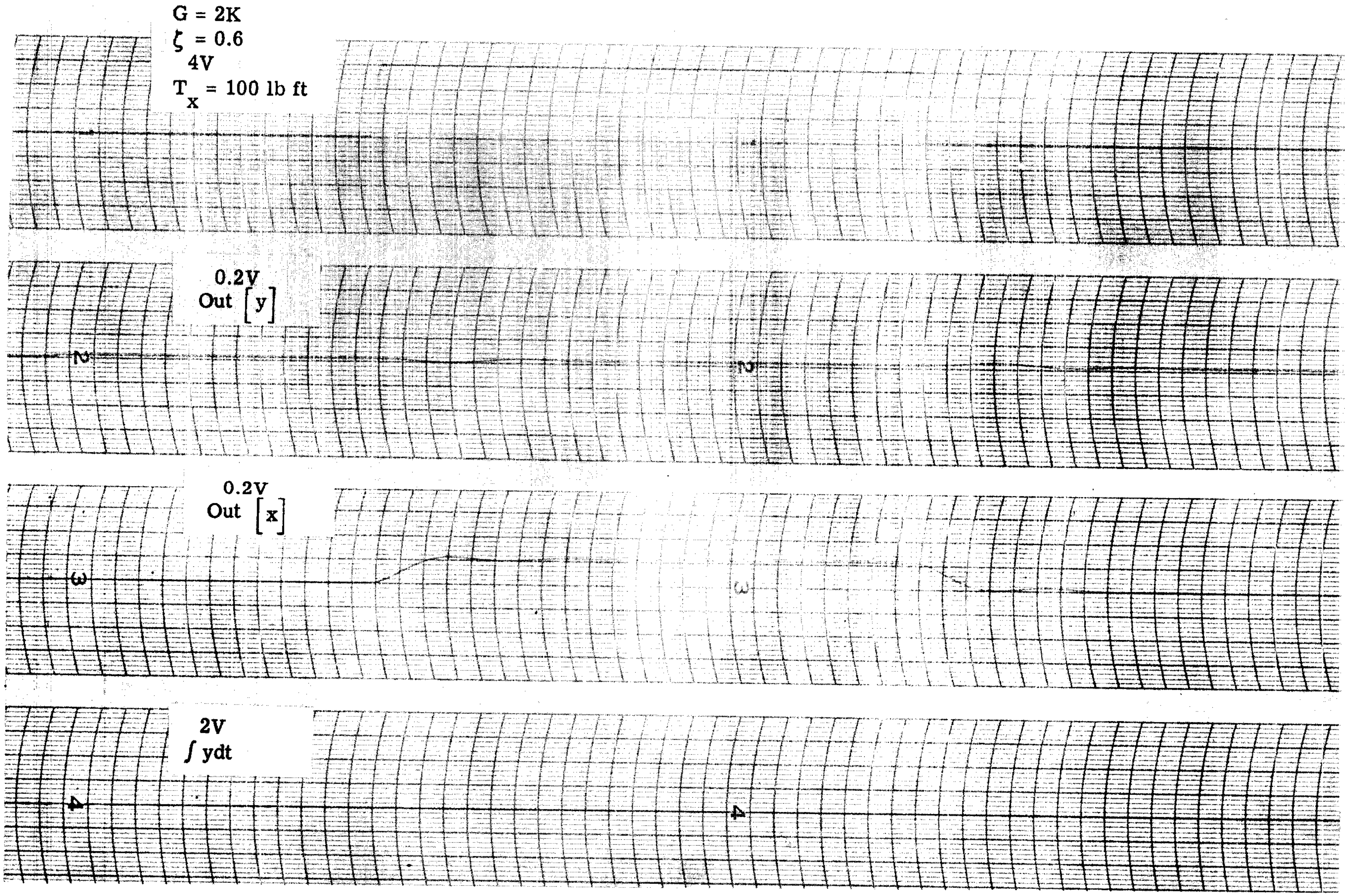


Figure IV-28. Response to 100 lb ft x Torque G = 2K lb ft/deg

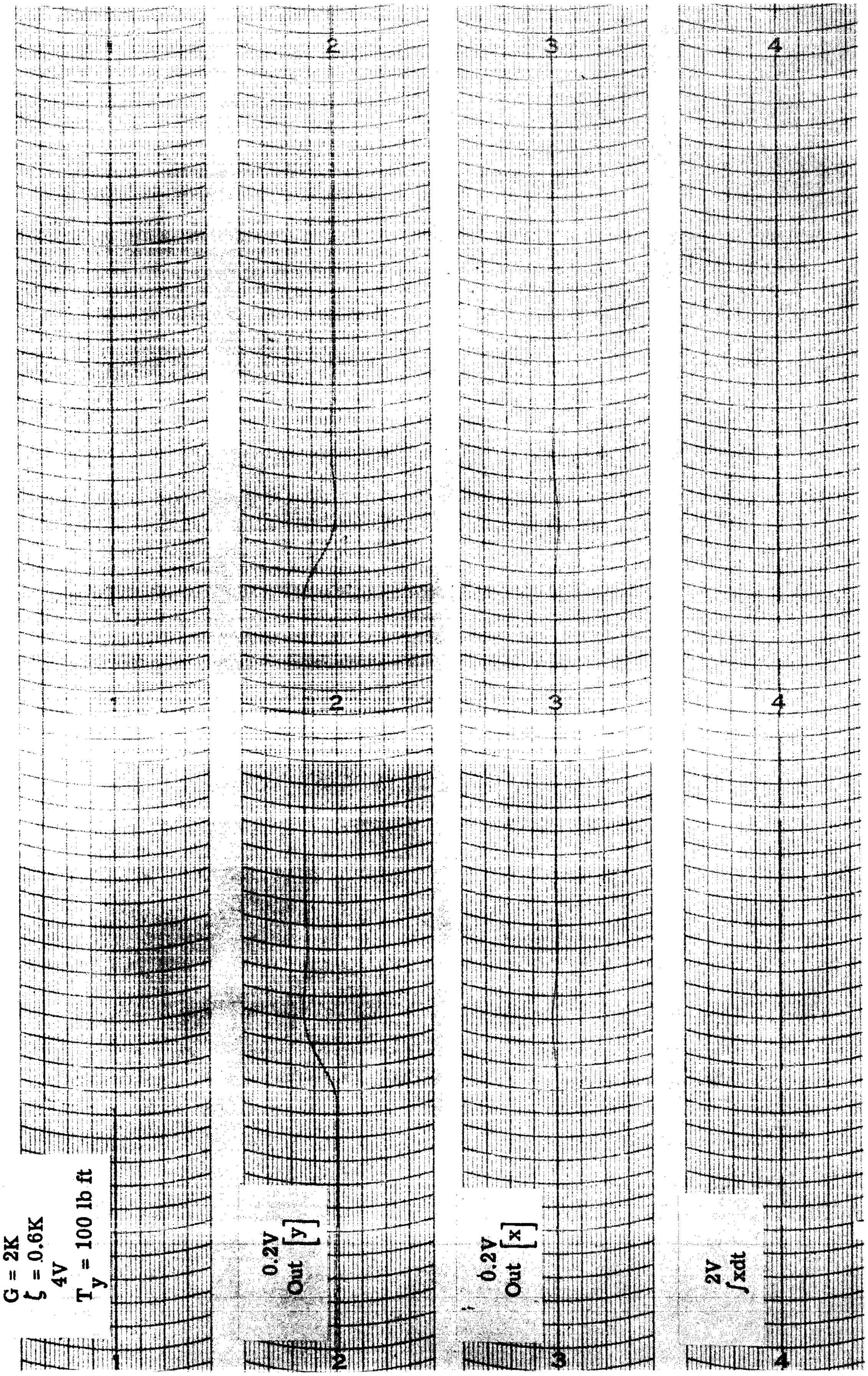


Figure IV-29. Response to 100 lb ft y Torque G = 2K lb ft/deg rigorous solution was obtained. The additional circuitry required for this solution is shown in Figure IV-30. The simulation of this solution was performed and it was verified that the axial rotation effects generated by the degree of assymmetry present were negligible.

The second consideration of axial rotation effects was to determine what results would occur if the vehicle were to be rotated in the presence of a wind. The proposed scheme does not allow freedom of axial rotation between the engine and the vehicle. Thus, effective angular momentum of the system would be greatly affected by any rotations of the vehicle, and commands generated by a wind direction sensor to keep the engine oriented in space might not be acted upon as effectively as with axial freedom. To find out what the results of this motion would be, sinusoidal inputs in quadrature were applied and the effective angular momentum was increased to correspond to the frequency of the inputs. Starting with very low rotational frequencies, where the outputs were identical to the inputs, the rotation was increased to greater than one radian per second. At this point the only effect was a thirty percent reduction in the vertical angle and a twelve degree displacement of the engine axis in the direction of the vehicle rotation. For a maximum vertical angle of ten degrees this is not serious.

SAFETY AND RELIABILITY

To determine factors of safety and reliability an additional eight simulations were performed. The results of these appear in Figures IV-31 to IV-38. They consist of various combinations of doubling or dividing by two, either the gain or the damping, for both x and y inputs of ten volt steps. Doubling the gain or dividing the damping by two does not result in an unstable system and dividing the gain by two or doubling the damping still allows the system to respond to the step within one second.

Rotation of the vehicle would tend to influence the gyroscopic effects rather markedly. This might be a source of instability; however, the tests performed illustrated that under the worst conditions that may occur, that the engine be called to maintain a ten degree tilt while the vehicle is rotating at better than one radian per second, no instability is present.

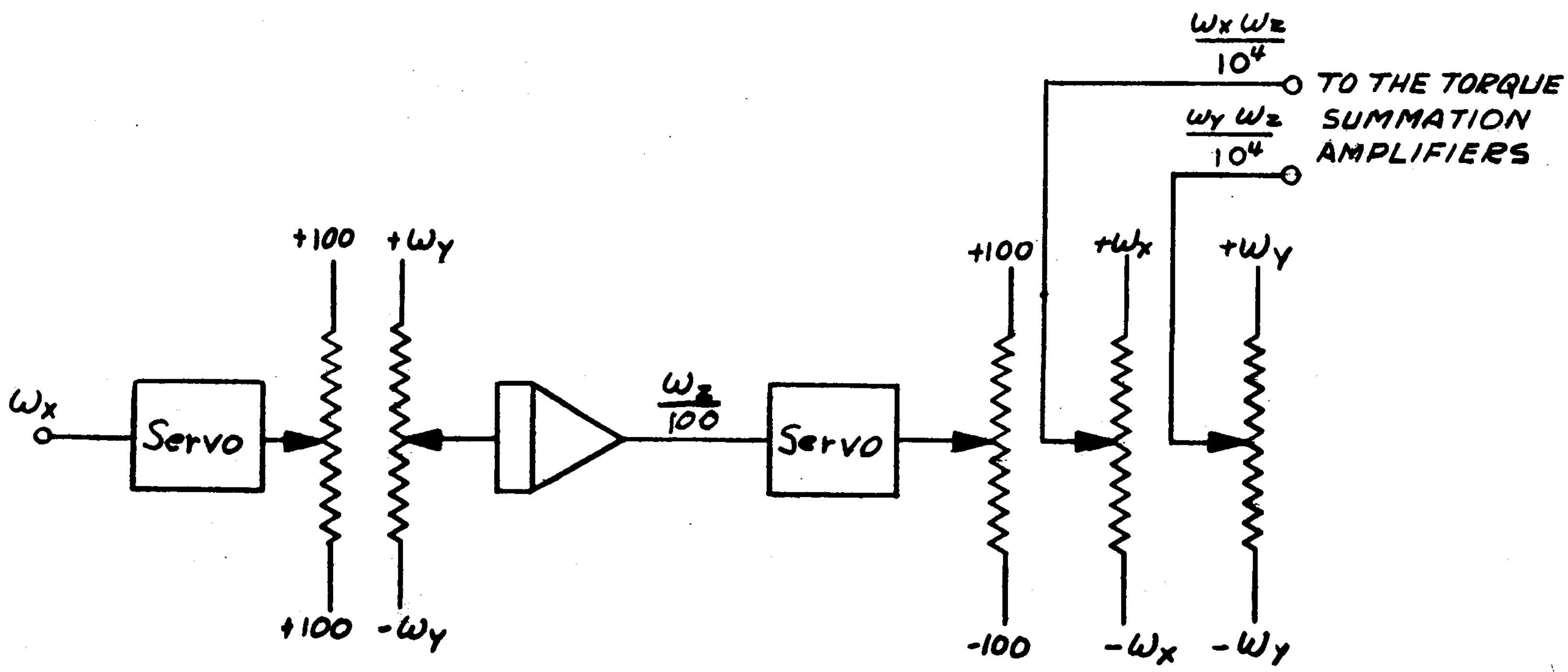


Figure IV-30. Analog Simulation Schematic

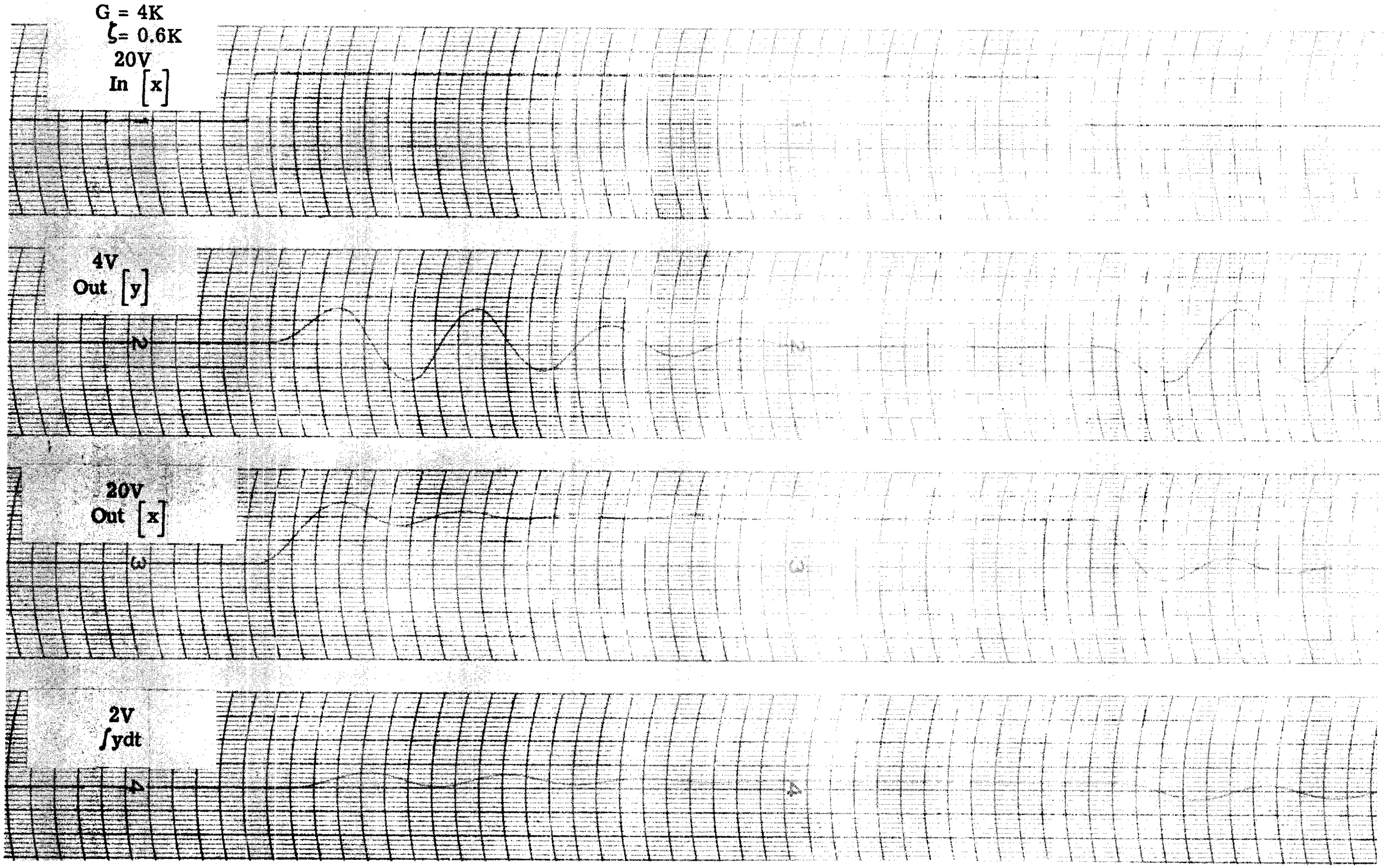


Figure IV-31. Response to 10° Step x Input, G = 4K lb ft/deg, ζ = 600 lb ft/deg/sec

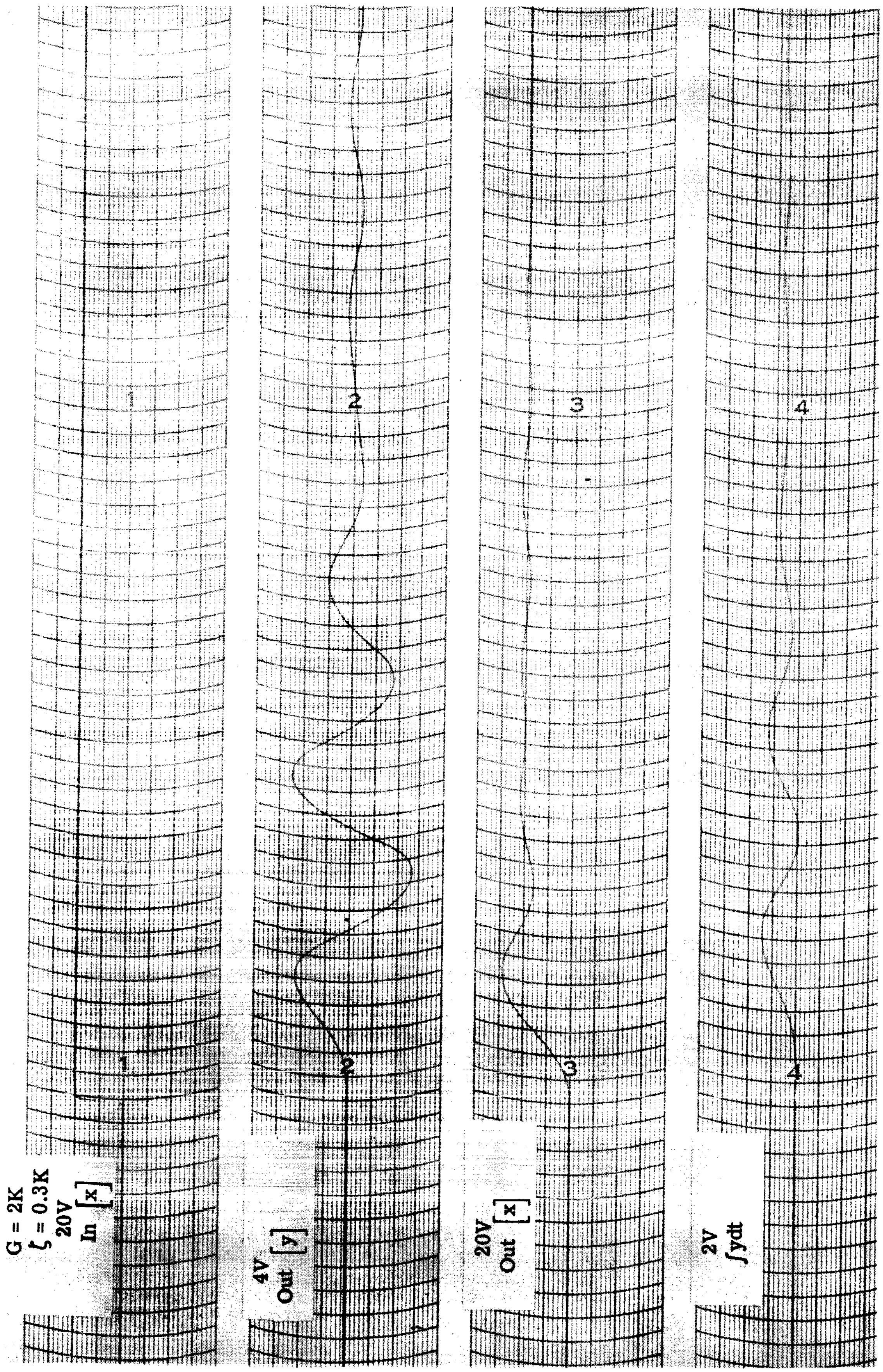


Figure IV-32. Response to 10° Step x Input, G = 2K lb ft/seg, ζ = 300 lb ft

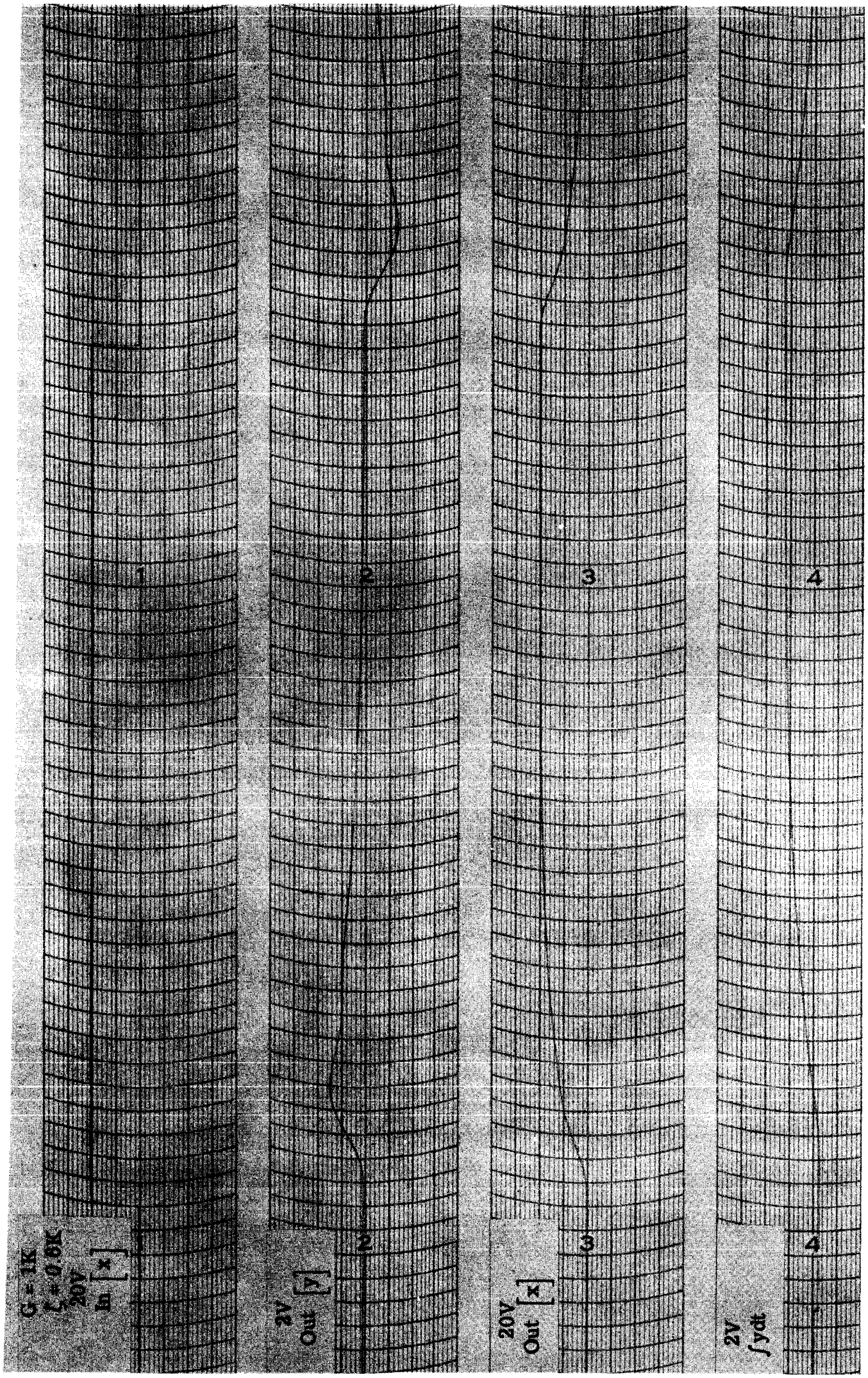


Figure IV-33. Response to 10° Step x Input, G = 1K lb ft/deg, ζ = 600 lb ft/deg/sec

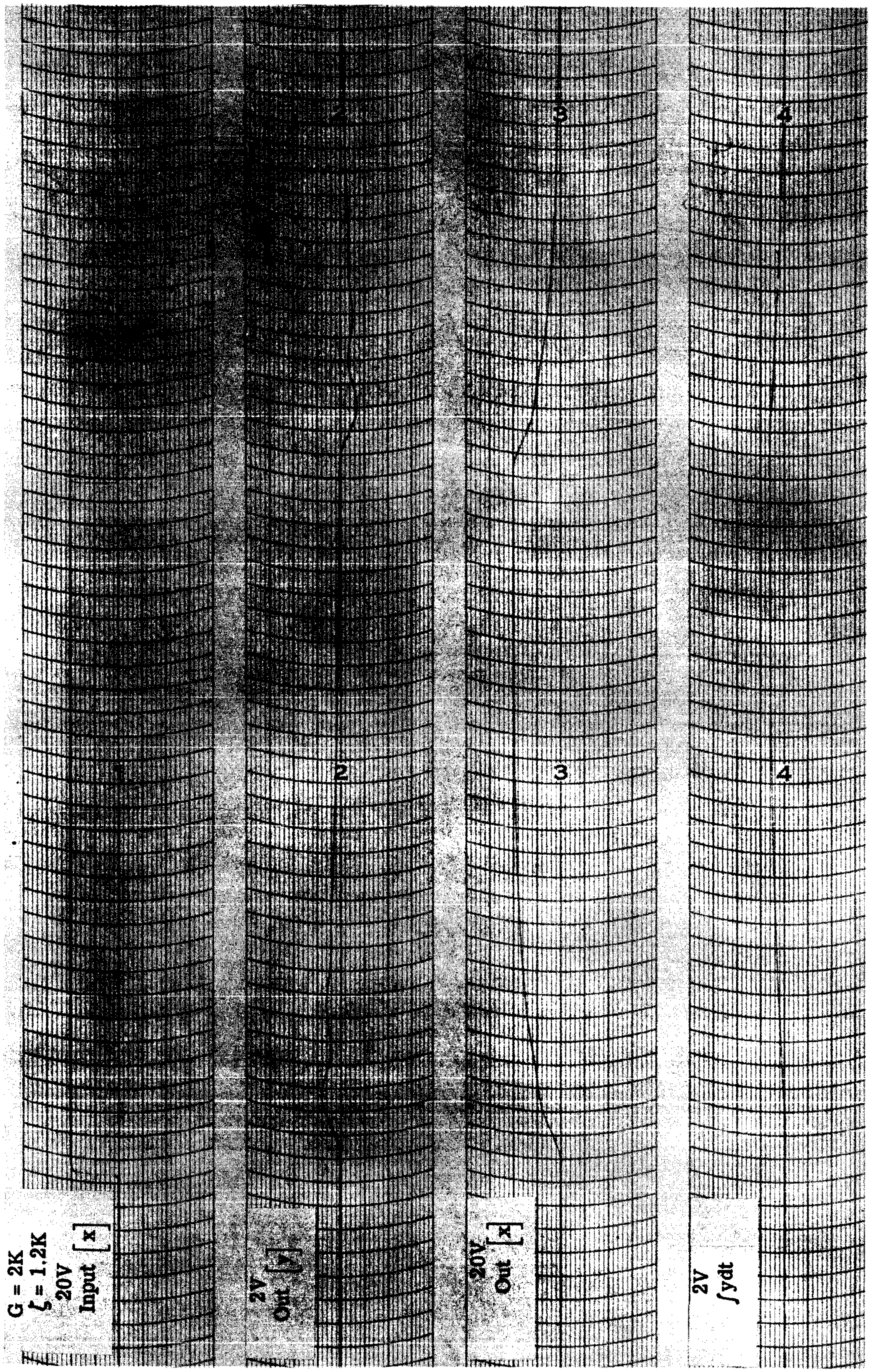


Figure IV-34. Response to 10° Step x Input G = 2K lb ft/deg ζ = 1.2K lb ft/deg/sec

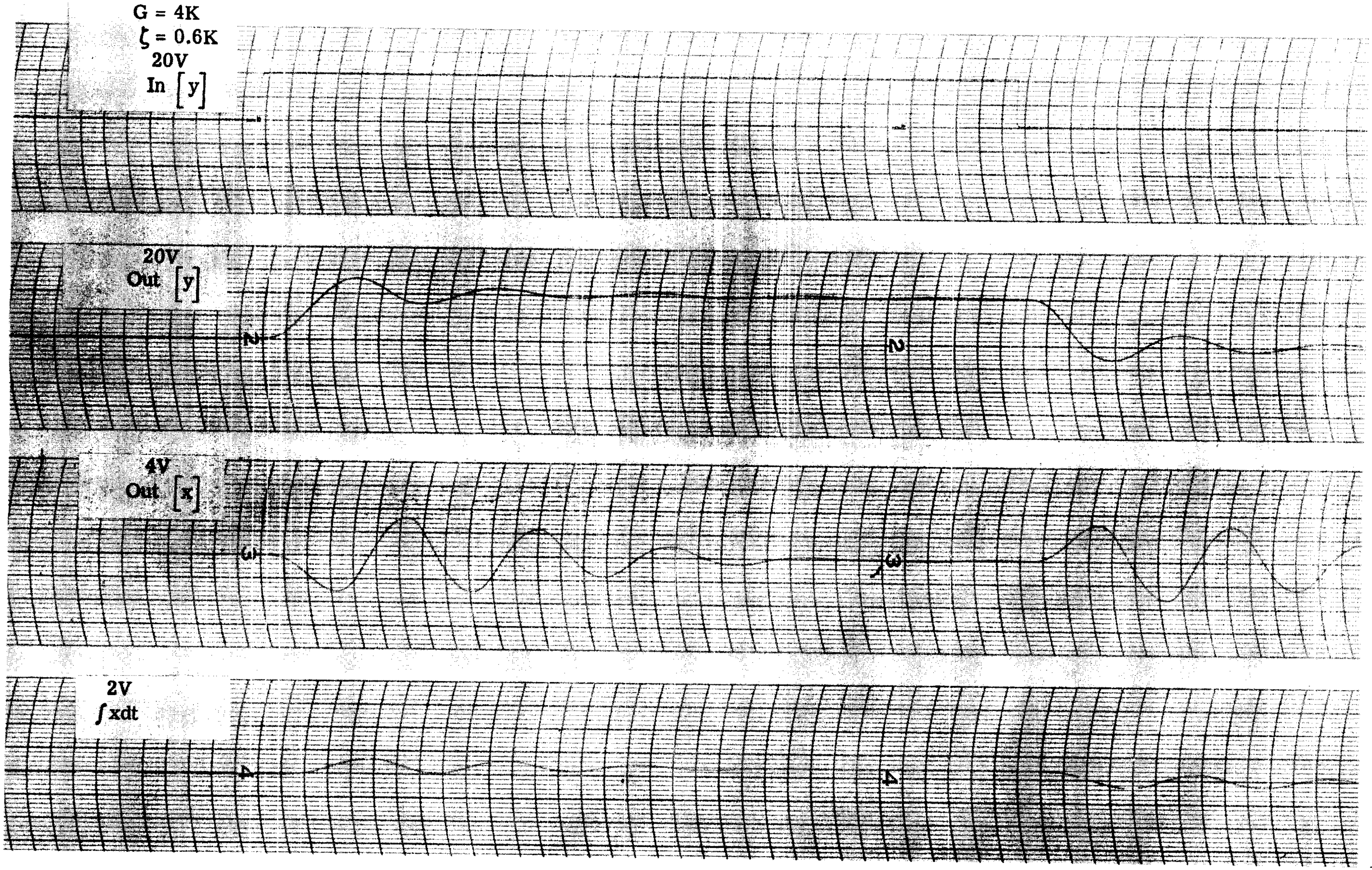


Figure IV-35. Response to 10° Step y Input, G = 4K lb ft/deg, ζ = 600 lb ft/deg/sec

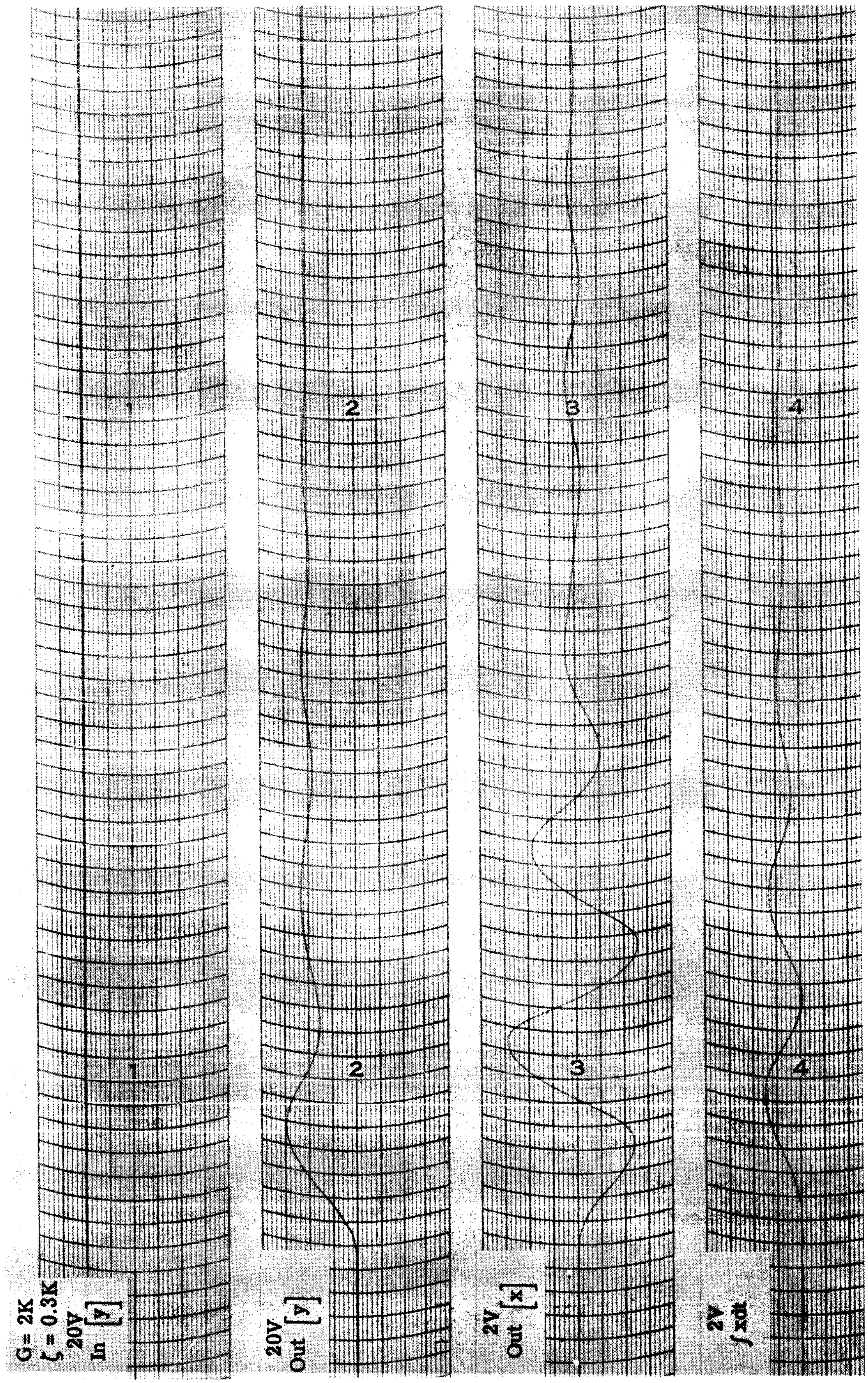


Figure IV-36. Response to 10° Step y Input, G = 2K lb ft/deg, ζ = 300 lb ft/deg/sec

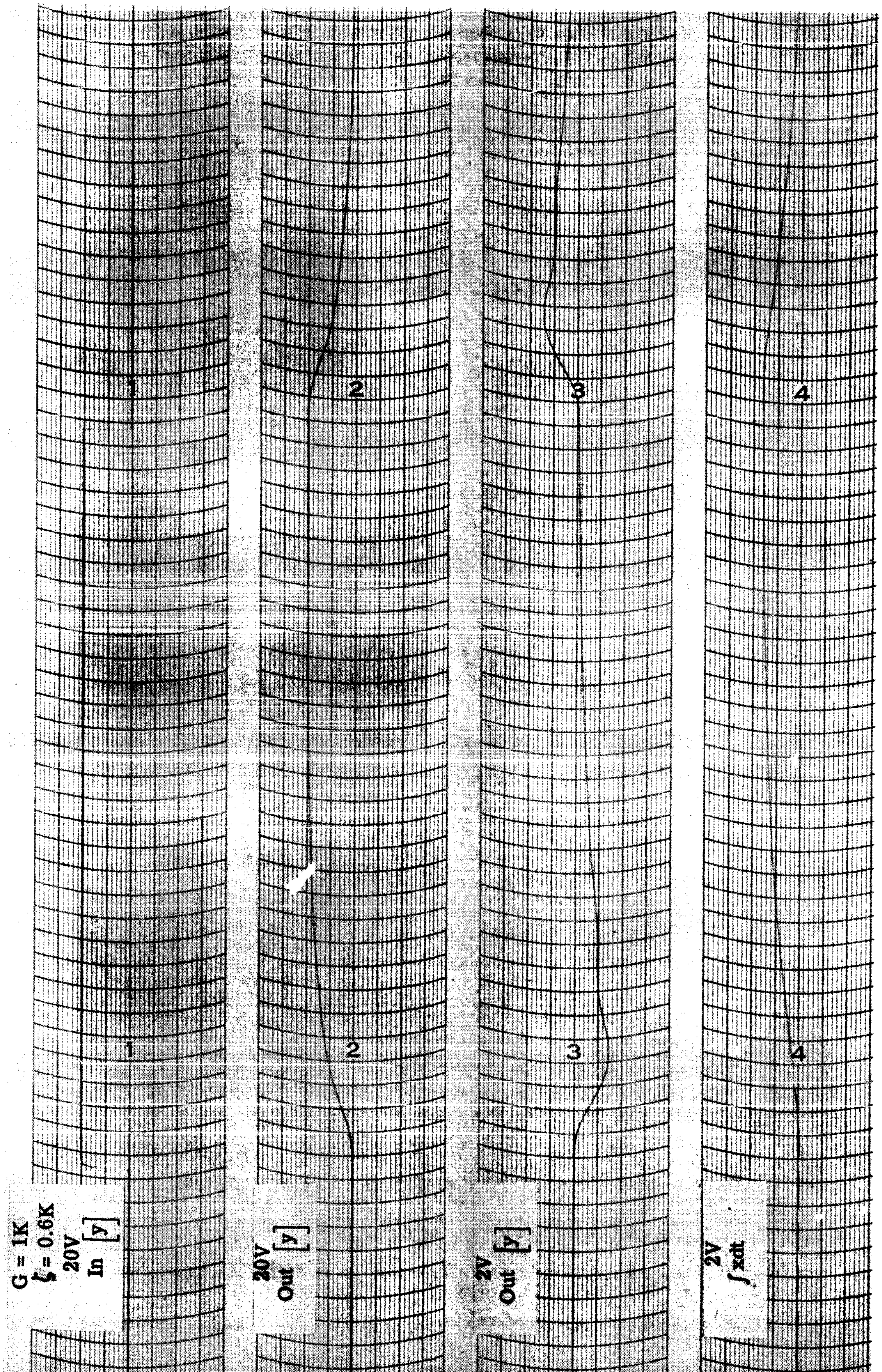


Figure IV-37. Responde to 10° Step y Input, G = 1K lb ft/deg, ζ = 600 lb ft/deg/sec

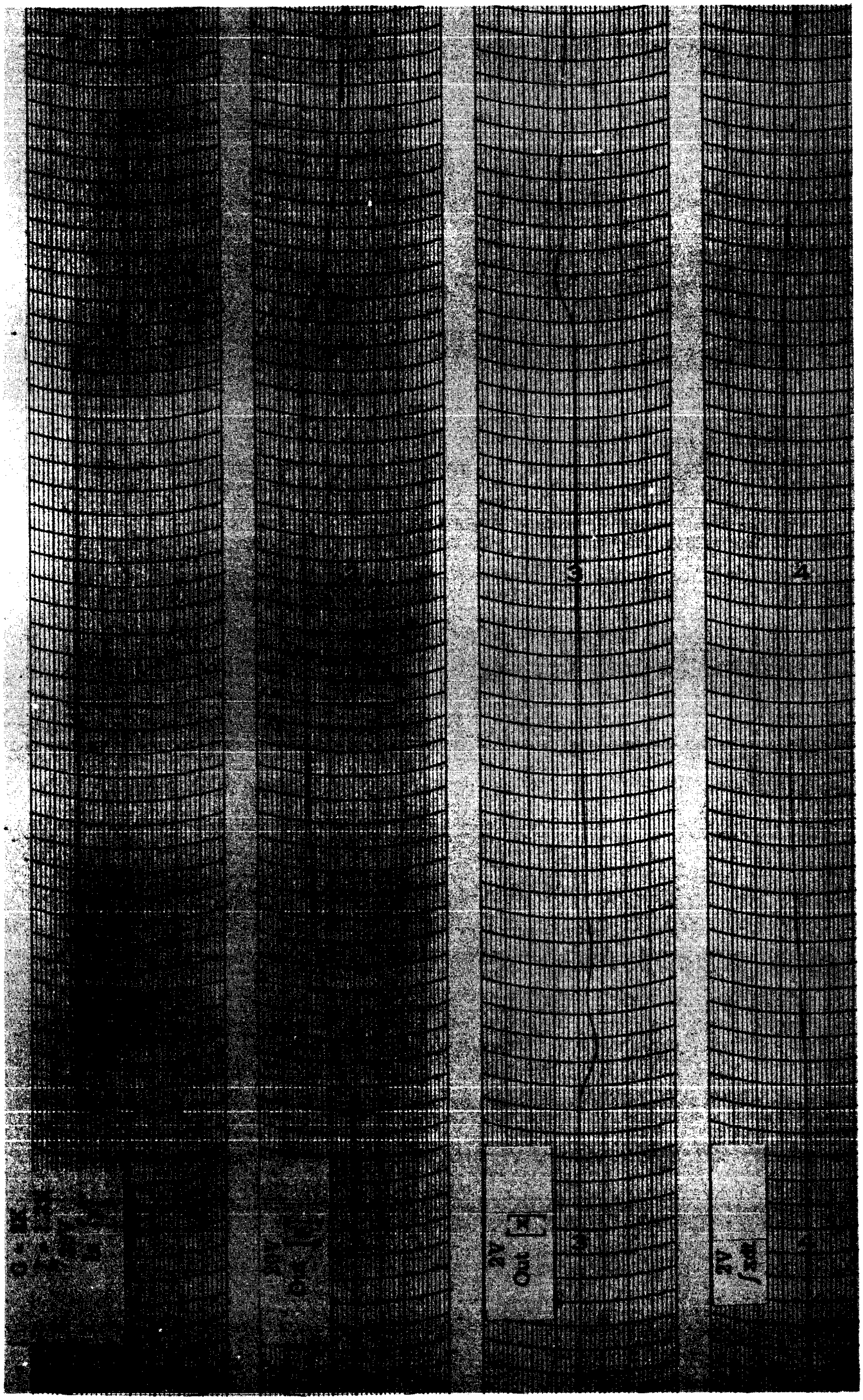


Figure IV-38. Responde to 10° Step y Input, G = 2K lb ft/deg, ζ = 1.2K lb ft/deg/sec

F. ANALOG COMPUTER SIMULATION OF VEHICLE

1. Description of Simulator and Tests

The Phase I analog simulation was designed to investigate attitude and translation control requirements for the free flight lunar landing simulator vehicle in the absence of aerodynamic forces and moments. The vehicle has been simulated in six degrees of freedom, under the influence of an earth gravity field, and zero density atmosphere.

The analog computers have been tied into a simulator cockpit to facilitate piloted control studies. The cockpit is equipped with a center stick and rudder pedals, a three axis side controller, a throttle which can be used to control either the lifting rockets or jet engine.

The simulator display comprises a pitch and roll attitude reference on a CRT (operates in same manner as a two axis gyro horizon), yaw indicator, dual scale altimeter reading from 0 to 400 ft or 0 to 4000 ft, rate of climb indicator reading ±4000 fpm, horizontal velocity projected display on a CRT, and a horizontal position display on a plotting board.

The analog simulation was used to investigate:

a. The range of pitch and roll control accelerations acceptable to the pilot and the optimum values for the nominal vehicle characteristics with neutral stability and no damping.

b. The range of c.g. travel above and below the gimbal plane which the pilot cannot detect, and second, the range which can be detected but is still acceptable. These studies were done with no damping and near optimum control power in pitch and roll.

c. Rocket engine thruster gradients were varied to determine the optimum attitude control response.

d. Stabilization of pitch and roll attitudes in the event of instantaneous thrust loss of one of the two lifting rocket engines.

e. Quantity of propellant required for attitude control.

The above studies were conducted using the center stick controller.

A very brief evaluation of the side controller was conducted. The results of this study are stated in the discussion of the analog study results.

The pilots for these studies were a Bell Aerosystems test pilot with flight test experience in the X-14 and Bell Air Test Vehicle, and a NASA test pilot.

2. Assumptions and Conditions

The vehicle is simulated in six degrees of freedom, with the moment equations written with respect to body axes and the force equations written with respect to earth axes.

a. The body axes are oriented with the origin at the center of gravity of the vehicle, the positive X-axis in the direction the pilot faces, the positive Y-axis through the support leg to the pilot's right, and the positive Z-axis downward and coincident with the principal axis.

b. The earth axes are oriented with the origin at a fixed point on the surface of the earth, the positive X-axis pointing north, the positive Y-axis pointing east, and the positive Z-axis toward the center of the earth.

c. The vehicle nominal characteristics are: weight of 3000 lbs, moments of inertia - I_X, I_Y, I_Z - are 2000 slug ft^2. Vehicle weight can be adjusted between 1500 lbs and 4000 lbs and the moments of inertia can be adjusted between 1000 and 3000 slug ft^2.

d. The jet engine thrust is set to five-sixths of the vehicle weight.

e. Lifting rocket thrust is throttleable from zero up to a maximum of 1/2 the vehicle earth weight, with capability for varying max thrust, idle thrust and throttle gradient independently.

f. The static margin - vertical center of gravity location with respect to the jet engine gimbal point - is adjustable through a range of ± 4 feet.

g. There is no transfer of jet engine dynamics due either to imperfect stabilization of gyroscopic torque.

h. All aerodynamic forces and moments are perfectly compensated. In lieu of the above, no natural damping exists; however, provision has been made for stability augmentation if desired.

i. Pitch, roll, and yaw control accelerations are generated as pure couples by the reaction control thrusters.

List of symbols and equations of motion are shown in Tables IV-2 and IV-3.

The simulation setup used on a Reeves REAC computer is shown in Fig. IV-39 and the potentiometer settings shown in Table IV-4.

TABLE IV-2

LIST OF SYMBOLS

g_e	earth referenced gravitational constant	ft/sec^2
h	altitude ~ equal to $(-)Z_e$	ft
I_X	moment of inertia about the body X axis	slug ft^2
I_Y	moment of inertia about the body Y axis	slug ft^2
I_Z	moment of inertia about the body Z axis	slug ft^2
K( )	angular damping gain	$\frac{lb\ ft}{rad/sec}$
l	static margin ~ vertical center of gravity location with respect to the jet engine gimbal point	ft
m	vehicle mass	slugs
L( )	moment about the body X axis ~ roll	lb ft
M( )	moment about the body Y axis ~ pitch	lb ft
N( )	moment about the body Z axis ~ yaw	lb ft
p	roll rate about the body X axis	rad/sec
q	pitch rate about the body Y axis	rad/sec
r	yaw rate about the body Z axis	rad/sec
T_J	thrust of the jet engine	lb
T_R	thrust of the lifting rocket engines	lb
W	vehicle weight	lb
W_J	jet engine weight	lb
W_{HP}	hydrogen peroxide consumption for attitude control	lb

TABLE IV-2 (Cont'd)

X_e	lateral displacement in the North-South direction	ft
Y_e	lateral displacement in the East-West direction	ft
Z_e	vertical displacement from the earth's surface	ft
δe	pitch control stick input	%
δr	yaw control rudder pedal input	%
δs	roll control stick input	%
θ	vehicle pitch attitude $\sim$ conventional Euler angle	deg
ϕ	vehicle roll attitude $\sim$ conventional Euler angle	deg
ψ	vehicle yaw attitude $\sim$ conventional Euler angle	deg
c	denotes control signal - used as a subscript	
o	denotes first derivative with respect to time when placed above a quantity	
oo	denotes second derivative with respect to time when placed above a quantity	

TABLE IV-3

LUNAR LANDING VEHICLE EQUATIONS OF MOTION

Earth Referenced Force Equations

$$m \ddot{X}_e = -T_R \left(\cos\psi \cos\phi \sin\theta + \sin\psi \sin\phi\right)$$

$$m \ddot{Y}_e = -T_R \left(\sin\psi \cos\phi \sin\theta - \cos\psi \sin\phi\right)$$

$$m \ddot{Z}_e = m g_e - T_J - T_R \left(\cos\phi \cos\theta\right)$$

Body Referenced Moment Equations

$$I_x \mathring{p} = (I_y - I_z)\, qr + L_c + \underset{=}{L_p}\overset{K_p p}{} - (T_J - W_J)\, l \sin\phi \cos\theta$$

$$I_y \mathring{q} = (I_z - I_x)\, rp + M_c + \underset{=}{M_q}\overset{K_q q}{} - (T_J - W_J)\, l \cos\phi \sin\theta$$

$$I_z \mathring{r} = (I_x - I_y)\, pq + N_c + \underset{=}{N_r}\overset{K_r r}{}$$

Gimbal Equations

$$\mathring{\phi} = p + \mathring{\psi} \sin\theta$$

$$\mathring{\theta} = q \cos\phi - r \sin\phi$$

$$\mathring{\psi} = (r \cos\phi + q \sin\phi) / \cos\theta$$

TABLE IV-4

POT SETTINGS - COMPUTER NO. 2

POT #	QUANTITY	FROM	TO	SCALED VALUE	GAIN	POT SETTING
1						
2						
3						
4	deg/rad	A.15	A.1	.573	1	.577
5	deg/rad	A.16	A.2	.573	1	.577
6	rad/deg	A.9	A.16	.035	1	.035
7	$\psi_o/200$	±100V	A.7	-	10	IC
8						
9	$Lc_{max}/10,000$	Trk 114	A.8		10	
10	$Mc_{max}/10,000$	Trk 116	A.10		10	
11	$Nc_{max}/10,000$	Trk 118	A.12		10	
12	$500/I_x$	A.8	A.4	500/2000	10	.265 nom
13	$500/I_y$	A.10	A.5	500/2000	10	.265 nom
14	$500/I_z$	A.12	A.6	500/2000	10	.265 nom
15	$K_p/2x10^4$	A.18	A.8	0	4	0 nom
16	$K_q/2x10^4$	A.19	A.10	0	4	0 nom
17	$K_p/2x10^4$	A.20	A.12	0	4	0 nom
18						
19						
20	$.50 \Delta \dot{p}(t)$	Fn.Sw.	A.4		1	
21	$.50 \Delta \dot{q}(t)$	Fn.Sw.	A.5		1	
22	$.50 \Delta \dot{r}(t)$	Fn.Sw.	A.6		1	
23	deg/rad	A.13	A.7	.573	1	.577
I.C. #						
1	$\theta_o/200$	±100V	A.1		10	IC
2	$\phi_o/200$	±100V	A.2		10	IC
3						
4						
5						
6						

Note: Those pot settings marked "nom" refer to the nominal vehicle configuration.

TABLE IV-4

POT SETTINGS - COMPUTER NO. 3

POT #	QUANTITY	FROM	TO	SCALED VALUE	GAIN	POT SETTING
1	$100/Ze_{max}$	A.7	A.1	.050	1	.050
2	$100/Ye_{max}$	A.6	A.2	.100	1	.100 nom
3	$100/Xe_{max}$	A.5	A.3	.100	1	.100
4						
5	$(I_y-I_z)/2500$	A.15	Trk 201		1	0 nom
6	$(I_z-I_x)/2500$	A.16	Trk 20		1	0 nom
7	$\mathring{Z}e_o/100$	$\pm$100 v.	A.7		10	IC
8						
9	$T_j x1/10,000$	A.8	Trk 204	2500/ 10,000	1	0 nom
10	$T_j x1/10,000$	A.18	Trk 203	2500/ 10,000	1	0 nom
11	10/M	A.13	A.7	10/93	1	.108 nom
12	10/M	A.11	A.5	10/93	1	.108 nom
13	10/M	A.12	A.6	10/93	1	.108 nom
14						
15						
16	$\Delta Z_e(t)/100$	Fn.Sw.	A.7		1	~
17	$\Delta Y_e(t)/100$	Fn.Sw.	A.6		1	~
18	$\Delta X_e(t)/100$	Fn.Sw.	A.5		1	~
19						
20	$K\dot{\omega}_{\phi}/\delta_s$	Trk 312	S.2 in		1	~
21	$K\dot{\omega}_{\theta}/\delta_e$	Trk 313	S.3 in		1	~
22	$K\dot{\omega}_{r}/\delta_r$	Trk 314	S.4 in		1	~
23	$(mg-T_J)/1000$	+100V	A.13	500/1000	1	.504 nom
I.C. #						
1	$Ze_o/2000$	±100V	A.1		10	IC
2	$Ye_o/100$	±100V	A.2		10	IC
3	$Xe_o/100$	±100V	A.3		10	IC
4						
5	$\dot{X}e_o/100$	±100V	A.5		10	IC
6	$\dot{Y}e_o/100$	±100V	A.6		10	IC

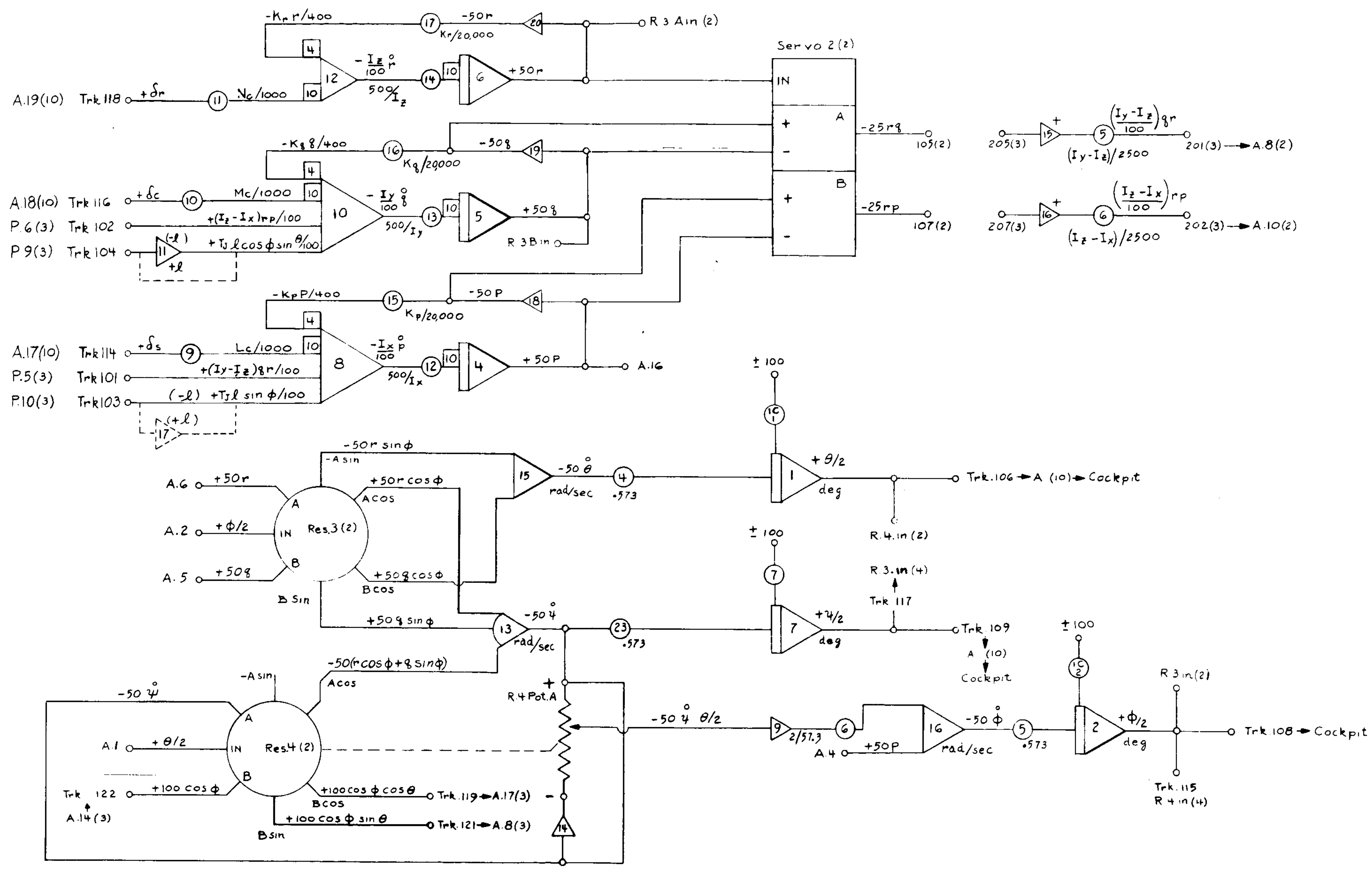
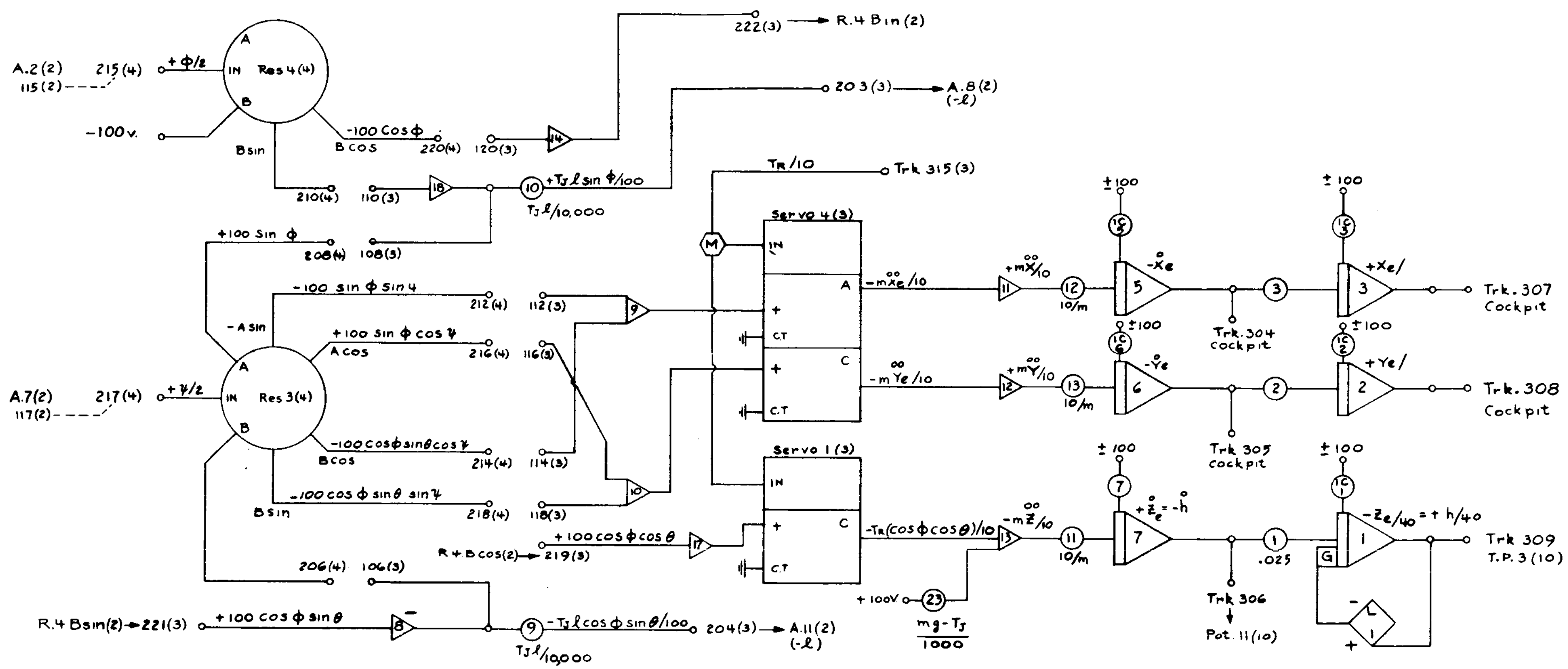


Figure IV-39. Analog Computer Mechanization



3. Discussion of Analog Study Results

The Cooper Pilot Opinion Rating System was used to evaluate the vehicle stability and control characteristics for the various configurations studied in this program. The Cooper System, which is described in Table IV-5, consists of rating numbers from 1 to 10, where a rating of 1 represents ideal characteristics and a rating of 10 represents catastrophic characteristics.

The pilot's mission assignment for all configuration studies was: Begin flight at an initial altitude of 2000 feet and 200 feet behind the desired landing site with the vehicle in equilibrium. Descend as rapidly as possible to 100 feet and bring the vehicle to momentary hover at that altitude. During the descent, translate forward until the vehicle is over the landing site. Descend to the surface making ground contact at a safe sink rate - less than 10 ft/sec - while controlling horizontal velocity to no more than 2 ft/sec at ground contact.

The pilot was given an initial familiarization period to "feel out" the vehicle with various combinations of vehicle restraint and for different stability configurations. After the commencement of a sequence of "mission" runs, no additional "free flight" was allowed.

For all configuration studies conducted in this program, the vehicle was controlled in five degrees of freedom. No yaw control was included. It is not felt however that the lack of yaw control seriously jeopardizes the results of this program, as lateral maneuvering can be accomplished through roll modulation, and thus the primary requirement for yaw control will probably be for rate stabilization.

The first program objective was to determine the acceptable range and optimum values of pitch and roll control powers for neutral static margin. The results are presented in Figure IV-40. The acceptable ranges and optimum values are:

	Maximum	Minimum	Optimum
Pitch Control Power	0.56 rad/sec^2	0.18 rad/sec^2	0.32 rad/sec^2
Roll Control Power	0.63 rad/sec^2	0.20 rad/sec^2	0.38 rad/sec^2

The minimum acceptable rating is defined as 4.5 on the Cooper scale. Referring to Figure IV-40, it can be seen that with the present simulation and instrument display, acceptable operation of the lunar landing simulator can be obtained. It should be expected

TABLE IV-5

COOPER PILOT OPINION RATING SYSTEM

Operating Conditions	Adjective Rating	Numerical Rating	Description	Primary Mission Accomplished	Can Be Landed
Normal Operation	Satisfactory	1	Excellent, includes optimum	Yes	Yes
		2	Good, pleasant to fly	Yes	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics	Yes	Yes
Emergency Operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics ~ marginal	Yes	Yes
		5	Unacceptable for normal operation	Doubtful	Yes
		6	Acceptable for emergency condition only ~ such as failure of stability augmentation	Doubtful	Yes
No Operation	Unacceptable	7	Unacceptable even for emergency condition	No	Doubtful
		8	Unacceptable - dangerous	No	No
		9	Unacceptable - uncontrollable	No	No
No Operation	Catastrophic	10	Motions possibly violent enough to prevent pilot escape	No	No

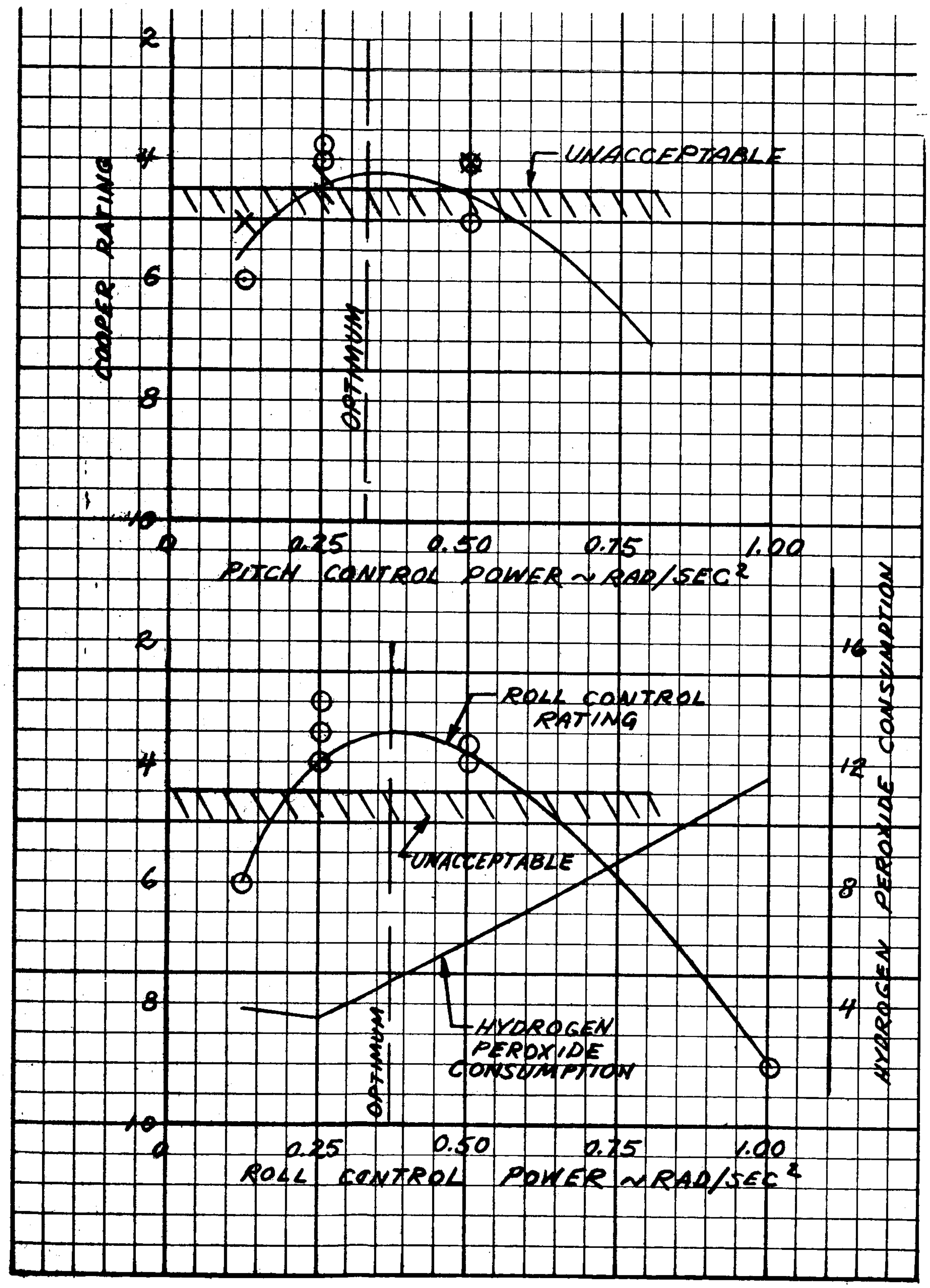


Figure IV-40. Pilot Evaluation of Vehicle Stability at Several Levels of Control Power that with further studies of displays and controls, even better results will be obtained.

The rate of hydrogen peroxide consumption of the attitude reaction controls is also shown in Figure IV-40. It appears that for reasonable values of pitch control, the peroxide consumption is primarily a function of roll control power. This was not unexpected, as lateral control is generally more gross in nature than is longitudinal control.

The maximum variations in static margin before vehicle stability becomes unacceptable are shown in Figure IV-41 for pitch and roll control powers of 0.25 rad/sec^2. This study indicated that the pilot was unable to detect any change in stability for center of gravity travel of at least 2 inches in either direction from the gimbal plane. Vehicle stability proved to be acceptable for at least 4 inches in either direction from neutral. Since the maximum center of gravity shift of the flight vehicle will not be more than 1 inch total, no appreciable deterioration in stability should be evident.

The flight times required to complete the basic mission with negative stability showed no appreciable deterioration with respect to flights made with neutral stability (flight durations ranged from 1:25 to 1:50 min/sec). Thus, even though the pilot's work load was increased as the static margin became more negative, it was possible to successfully complete the mission, even for a negative margin of six inches.

The throttleable thrust of the lifting rocket engines was varied between $T = 0$ to $W/3$ and $T = W/12$ to $3\ W/12$. The most sensitive altitude control, corresponding to a gradient of $(W/3)/100\%$ of throttle, appeared to be most satisfactory. When descending at a high sink rate, a fairly steep thrust gradient allows the pilot to bring the vehicle to hover with a very positive response and minimum lag.

In the event of instantaneous thrust loss of one of the lifting rockets, an angular acceleration of .5 rad/sec^2 would be acting on the nominal vehicle's outer frame. Several flights were made introducing an unbalance moment equivalent to an engine loss into either the pitch or roll equations. The duration of the disturbances was on the order of 1 second - assuming this to be sufficient time to shut down the opposing engine either automatically or manually. The pilot was consistently able to bring the vehicle under positive control in less than 5 seconds with no more than three overshoots. These studies were made with pitch and roll control levels of only .25 rad/sec^2.

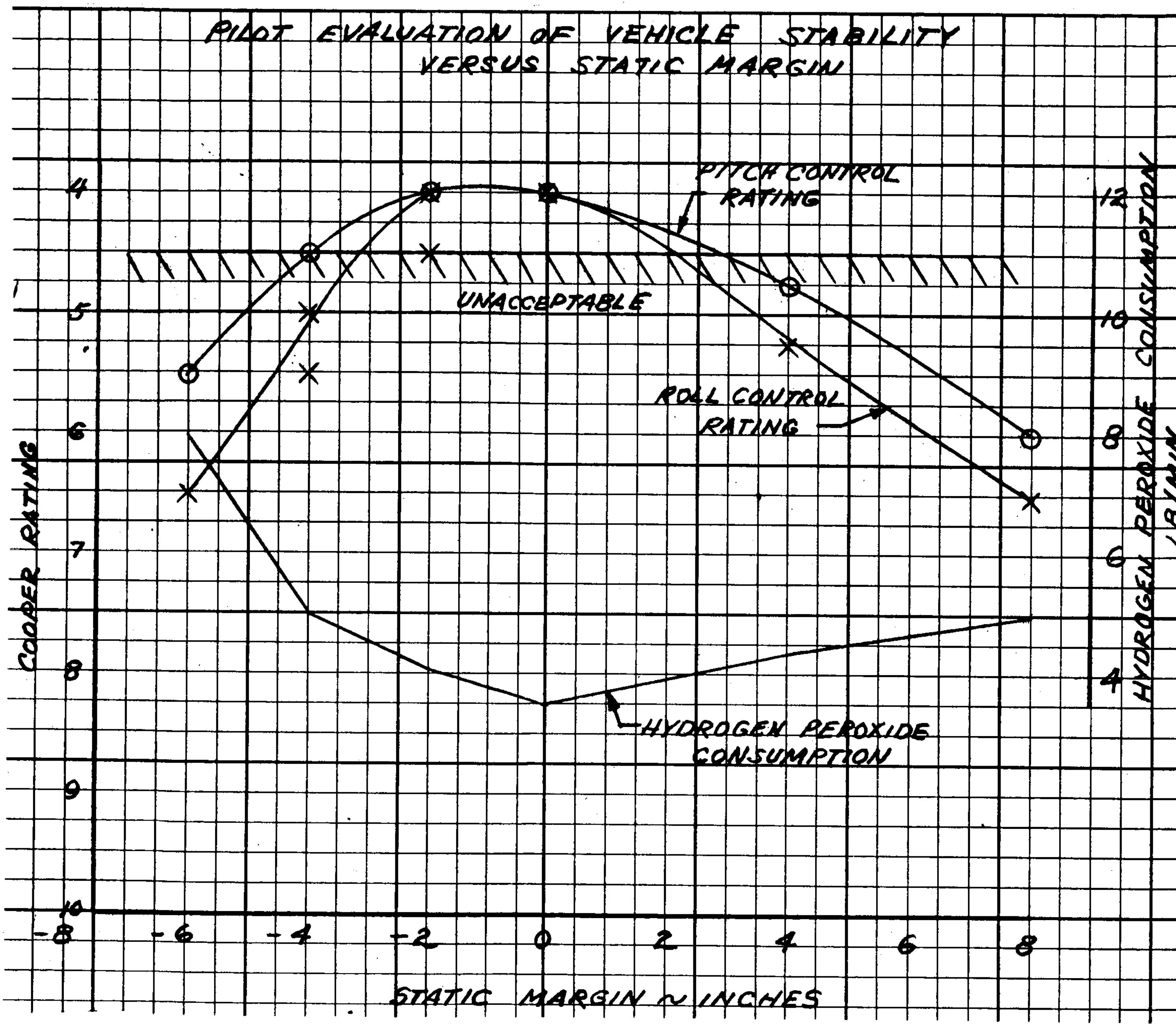


Figure IV-41. Pilot Evaluation of Vehicle Stability versus Static Margin

A three axis side controller on loan from NASA Flight Research Center at Edwards Air Force Base has been installed in the cockpit simulator. This controller has rotation limits of $\pm 30^{\circ}$ in pitch and yaw and $\pm 25^{\circ}$ in roll. A spring-cam system for force feel and centering is used in all three axes with the spring tensions being adjustable.

It appears that for an acceleration control system where high frequency/high magnitude inputs are required, the continual wrist swiveling becomes quite fatiguing. The fatigue factor aggravates the problem of cross controlling and lends to gradual deterioration of pilot control sensitivity. This first evaluation of the controller was concerned only with the pitch and roll axes. With the addition of yaw control, the fatigue and cross coupling problems will become even worse.

The performance of the present controller could be improved somewhat by modification of the spring cam systems and pitch counter-balancing; however, a better approach (for a side controller) may be a small pencil type stick with either force or displacement outputs. The decision as to the optimum type attitude controller must be investigated.

4. Future Simulation Expansion

The next step in analog computer simulation is based on six degree of freedom vehicle and six degree of freedom jet engine analog mechanizations. The vehicle simulation should include:

(a) Generation of aerodynamic forces and moments

(b) Mechanization of the dynamics of the landing struts

(c) Dynamic coupling to the jet engine

Capability will exist to program vehicle weight, moments of inertia, and center of gravity location with rate of fuel burnoff.

The jet engine simulation will include:

(a) Jet engine dynamics including the aero effects on the engine

(b) Jet engine stability augmentation and aerodynamic compensation systems

A new cockpit simulator is recommended with flexibility for the installation of:

(a) Center stick and side controllers with variable dynamics

(b) Several rocket and jet engine throttle configurations

(c) Quick change instrumentation and display packages

(d) A system for generating visual cues

The analog simulation will facilitate the study of:

(a) Basic pilot control problems

(1) Investigate attitude control power and damping requirements for various vehicle configurations

(2) Throttle control of the rocket and jet engines in all flight control modes

(b) Operation in all parts of the flight envelope with any of the primary or emergency control modes. This capability will allow:

(1) Investigation of and optimization of a variety of flight profiles

(2) Maximization of the flight envelope

(3) Evaluation of emergency control procedures

(4) Landing sutides including touchdown on inclines with horizontal translation

Simulation of Aerodynamic Forces and Moments

Analog computation of aerodynamic forces and moments acting on a VTOL vehicle such as the free flight lunar landing simulator presents a much more formidable problem than in the case of conventional aircraft. The primary reason for this is that angles of attack and sideslip can vary quite rapidly through large angles - sometimes as much as 180° - in and around hovering flight, and become indeterminate at the over point. It is therefore evident that an attempt to mechanize the aerodynamic forces and moments on the basis of α and β will lead to operational difficulties if servo multipliers and resolvers are driven by these quantities.

The disappearance of the relative wind at hover makes it undesirable to use any coordinate system based on its direction. Thus, wind and stability axes are eliminated from consideration.

By generating the aerodynamic forces and moments with respect to body axes and referring the relative wind velocity components to these axes, it is possible to mechanize functions on the computers which are continuous and slowly varying at and around hover. One method of generating the forces and moments with respect to body axes is:

(a) The aerodynamic quantities are arrived at through wind tunnel tests and dimensionalized with respect to body axes.

(b) The dimensional quantities are then approximated in polynomial form as functions of the body referenced relative wind vectors. The fitting procedure is accomplished through application of an IBM least squares approximation program.

Based on the above procedure, the aerodynamic force in the direction of one of the body axes might take the form:

$$
\begin{aligned}
F(\ )_{body} &= f(\dot{X}_b, \dot{Y}_b, \dot{Z}_b) \\
&= C_1\dot{X}_b + C_2\dot{X}_b^2 + C_3\dot{X}_b\dot{Y}_b + C_4\dot{Y}_b + C_5\dot{Y}_b^2 \\
&\quad + C_6\dot{X}_b\dot{Z}_b + C_7\dot{Z}_b + - - -
\end{aligned}
$$

(including as many terms as needed to get the desired accuracy)

Considering the symmetric configuration of the lunar landing vehicle with respect to the X,Z and Y,Z body reference planes, simplification of the above method may be applicable.

Resolve the aerodynamic force acting on the vehicle into two components: one in the direction of the Z body axis; and the other in the direction of the free air velocity vector projected into the X,Y body plane. These two forces can be generated as functions of two variables, $(\dot{X}_b^2 + \dot{Y}_b^2)^{1/2}$ and $\dot{Z}_b$.

(let $(\dot{X}_b^2 + \dot{Y}_b^2)^{1/2}$ be represented by $\dot{X}$)

The form of these forces would be

$$F_{x,y} = f(\dot{X}, \dot{Z}_b)$$

$$= C_1\dot{X} + C_2\dot{X}^2 + C_3\dot{X}\dot{Z} + C_4\dot{Z}_b + ---$$

and

$$F_z = g(\dot{X}, \dot{Z}_b)$$

$$= K_1\dot{X} + K_2\dot{X}^2 + K_3\dot{X}\dot{Z} + K_4\dot{Z}_b + ---$$

Resolving $F_{x,y}$ into components in the X_b and Y_b directions gives

$$F_x \cong F_{x,y}\,(\dot{X}_b/\dot{X})$$

and

$$F_y \cong F_{x,y}\,(\dot{Y}_b/\dot{X})$$

Approximations of the above type should be sufficiently accurate for studies in the velocity range of a lunar landing vehicle.

V. VEHICLE AND ENGINE PERFORMANCE

The engine selected as the auxiliary lift device is a General Electric CF-700-2B. It is an axial-flow fan-jet engine, using the same gas generator as the GE J-85 in combination with a free floating, single-stage aft fan.

The basic performance of this engine was obtained from References 1 and 2. Data used in the vehicle performance analysis is presented in this section.

A. ENGINE PERFORMANCE

1. Basic Performance

Thrust and fuel flow of the CF-700-2B engine is presented in Figures V-1 to V-4 for the range of flight conditions expected. Standard Day parameters were obtained directly from Reference 1. Warm Day (Standard Day plus 27°F) thrust and fuel flows were obtained by extrapolating the Standard Day data using Reference 2 as a guide. (Note: Reference 2 is a complete performance manual for the CF-700-1 engine, the first of the CF-700 series. The CF-700-2B offers approximately 5% better thrust and SFC values on a standard day, and, although the data is incomplete, appears to give about 10% better warm day performance than the CF-700-1.) Correction factors due to compressor air bleed were applicable and were used directly from Reference 2.

The bleed air from the compressor is used in control jets to stabilize and control the engine during lunar simulation and hence, produces a small amount of useful thrust. The magnitude of this thrust ranges from 100 to 150 lbs, for the case of 6% bleed. This control jet thrust is included in the thrust numbers whenever bleed is specified.

Reference 1 General Electric CF-700-2B Aft Turbofan Engine Performance Notebook, 11 August 1961.

Reference 2 General Electric CF-700 Turbofan Engine Performance Bulletin, April 1960.

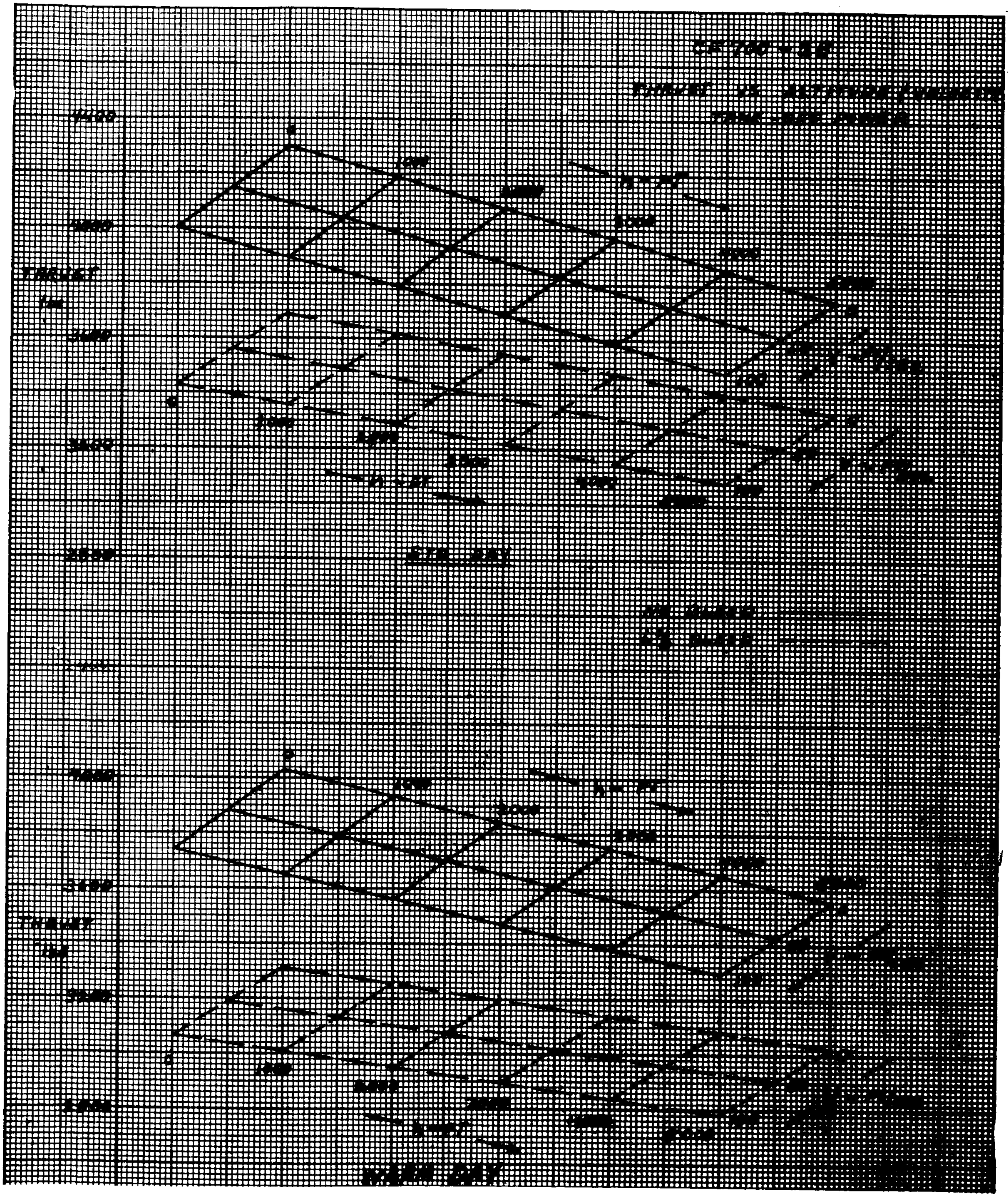


Figure V-1. Thrust versus Altitude and Velocity Takeoff Power

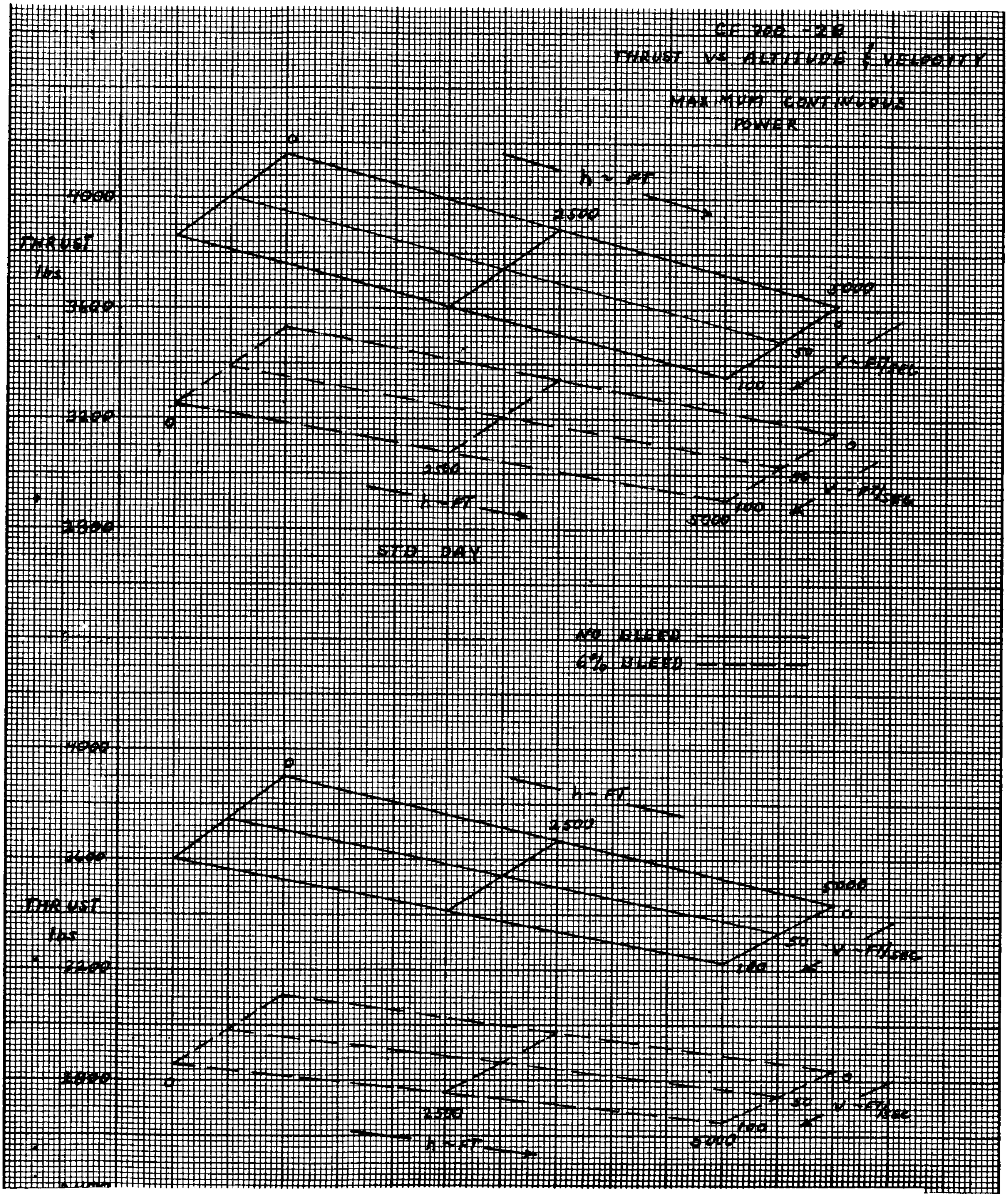


Figure V-2. Thrust versus Altitude and Velocity

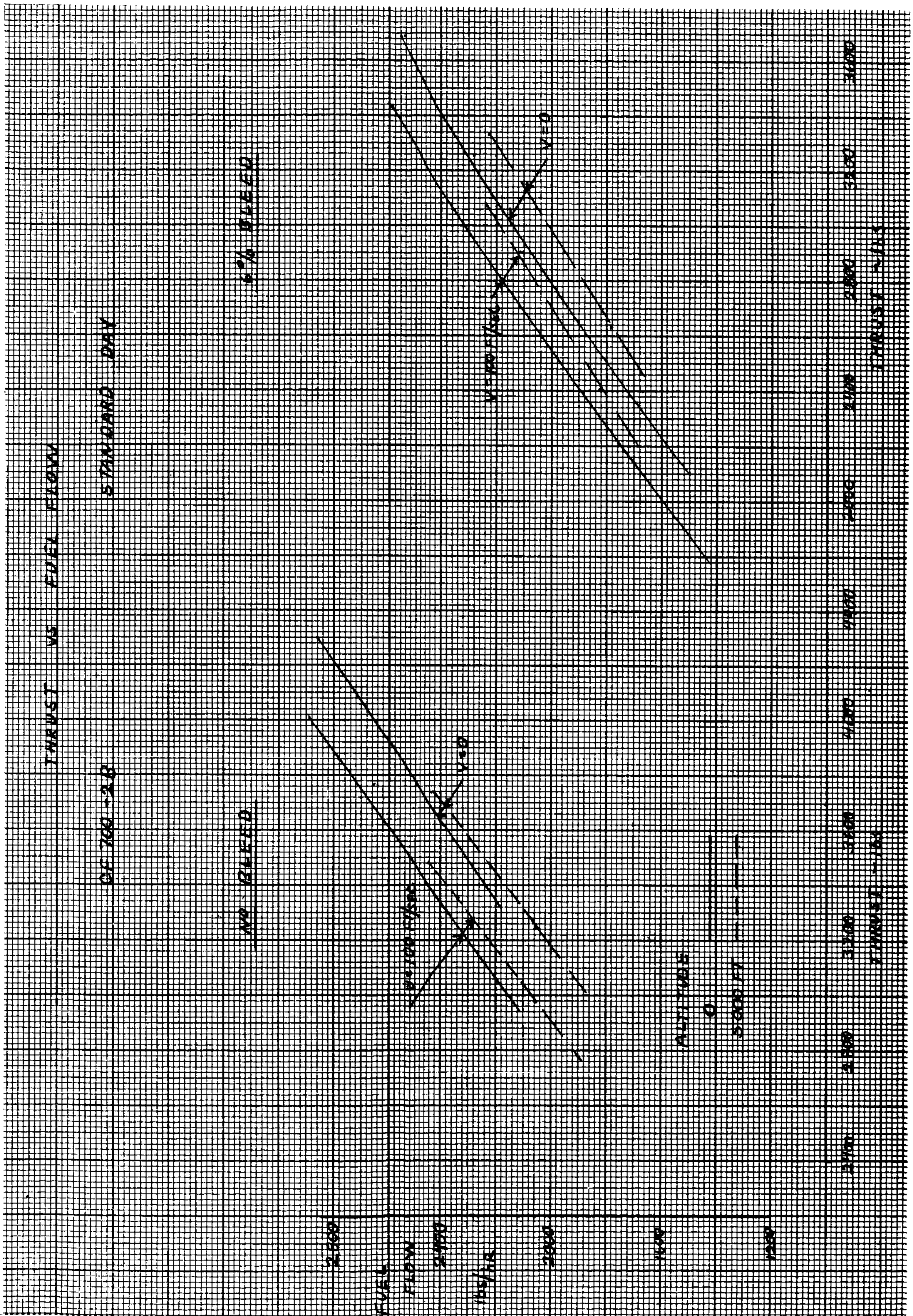


Figure V-3. Thrust versus Fuel Flow

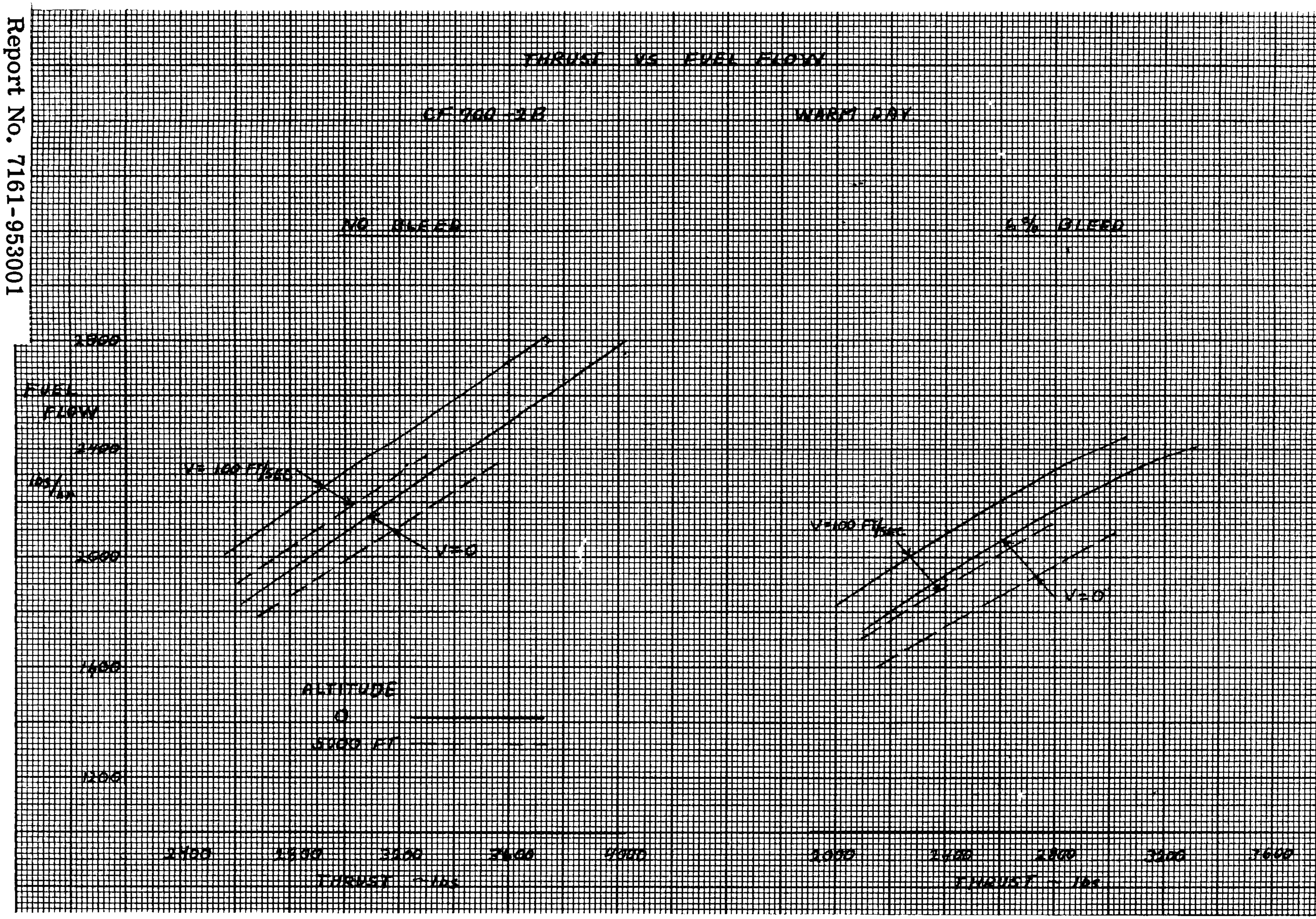


Figure V-4. Thrust versus Fuel Flow

2. Installation

The installation of the CF-700-2B in the present configuration is not expected to affect the basic engine performance. The presence of the capsule floor near the inlet and the structure surrounding the engine will create some disturbances in the flow field. These effects on engine performance will depend both on speed and direction (i.e. angle of attack). The magnitude of the effects will have to be determined from tests, but, considering the relatively low (less than 100 ft/sec) velocities, should cause no serious performance losses. Static tests conducted by General Electric on a J-85 indicate no loss in performance with a flat plate located 1/2 inlet diameter (dia. = 16 in.) in front of the air inlet bell mouth. Clearance on this vehicle is over one diameter.

3. Vertical Descent

The problems of vertical descent at speeds of up to 100 ft/sec were discussed with both General Electric and Bell Aerosystems personnel. There are two possible effects of "flying backwards". One is the effect on thrust at negative velocities; the other is the possibility of hot exhaust gas recirculation. Although there are no test data or analyses available on these problems, experienced people from both General Electric and Bell Aerosystems felt that there would be no detrimental effects. Indeed, from the shape of the thrust vs velocity curves, one should expect a slight increase in thrust at velocities below zero (i.e. backwards). The recirculation problem is discussed elsewhere in this report. It was assumed for the performance analysis, that there were no effects on engine performance.

B. VEHICLE PERFORMANCE

The basic performance of the complete vehicle is presented in this section. Included are the operating envelopes and curves defining the maximum vertical and lateral speed. These limits are set by the jet engine, which is the source of both the simulated gravity field and the force which nullifies the aerodynamic drag.

The following ground rules were used in the performance calculations:

1. Thrust and fuel flow are presented in Figures V-1 to V-4. It is assumed that the thrust at negative velocities (i.e. descending flight) is equal to the values at V = 0.

2. Takeoff gross weight is 3400 pounds. Jet engine fuel weight is 400 pounds.

1. Takeoff and Climb

Normal takeoff and climb to mission altitude is accomplished with the engine gimbals caged and using the "takeoff" power setting. Control is by means of the vehicle attitude control system and therefore does not require compressor bleed air from the engine.

Figure V-5 shows thrust available and thrust required (drag and weight) for vertical operation between 2000 and 4000 feet. Table V-1 summarizes the time and fuel required to climb from 2000 to 4000 feet.

Figure V-8 shows the operating envelopes for vertical ascending flight for a gross weight of 3400 lbs.

2. Translational and Hovering Flight

During mission simulation the jet engine is throttled and vectored to provide a vertical component of thrust equal to five-sixths of the vehicle weight and the force necessary to overcome the total aerodynamic force acting on the vehicle. The aerodynamic force consists mainly of a drag force opposite in direction to the velocity.

The engine attitude is controlled using compressor bleed air. For purposes of the performance calculation, a constant six per cent bleed is assumed. The thrust and fuel flows shown in Figures V-1 to V-4 include the effect of bleed on the engine performance and the thrust recovered through use of the bleed air. Figure V-6 shows the thrust required as a function of vehicle weight and velocity. Cross plotted on these curves are the thrust available at maximum continuous power and lines of constant engine deflection. It can be seen that the thrust required depends only slightly on the velocity; the limiting factor in lateral velocity is the engine deflection angle. This angle, for lateral flight, is shown in Figure V-7 for a range of vertical thrust components (i.e., weight supported). Using the thrust required curves from Figure V-6, the vehicle flight time can be computed. Table V-2 summarizes endurance in hovering and lateral flight. The operating envelope of the vehicle during lunar simulation is shown in Figure V-8. This is based on the maximum continuous power of the engine supporting a gross weight of 2833 lbs, (i.e., 5/6 of 3400).

A typical standard day mission, assuming a climb to 2000 feet and a simulation period of 10 minutes, requires 338 pounds of fuel (43 lbs. at R/C = 30 ft/sec. + 295 lbs for 10 minutes operation) and leaves 62 lbs as a safety factor.

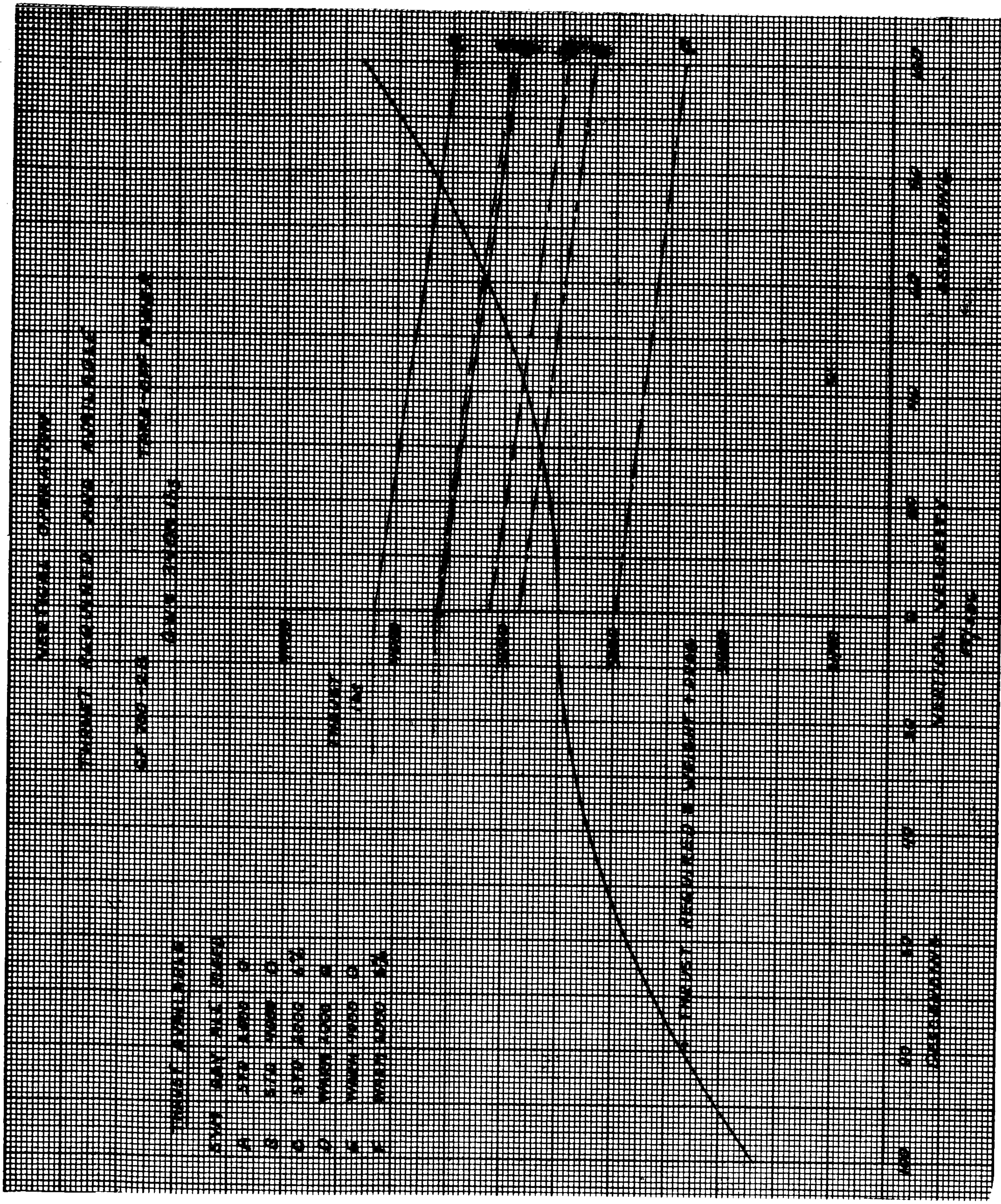


Figure V-5. Vertical Operation Thrust Required and Available

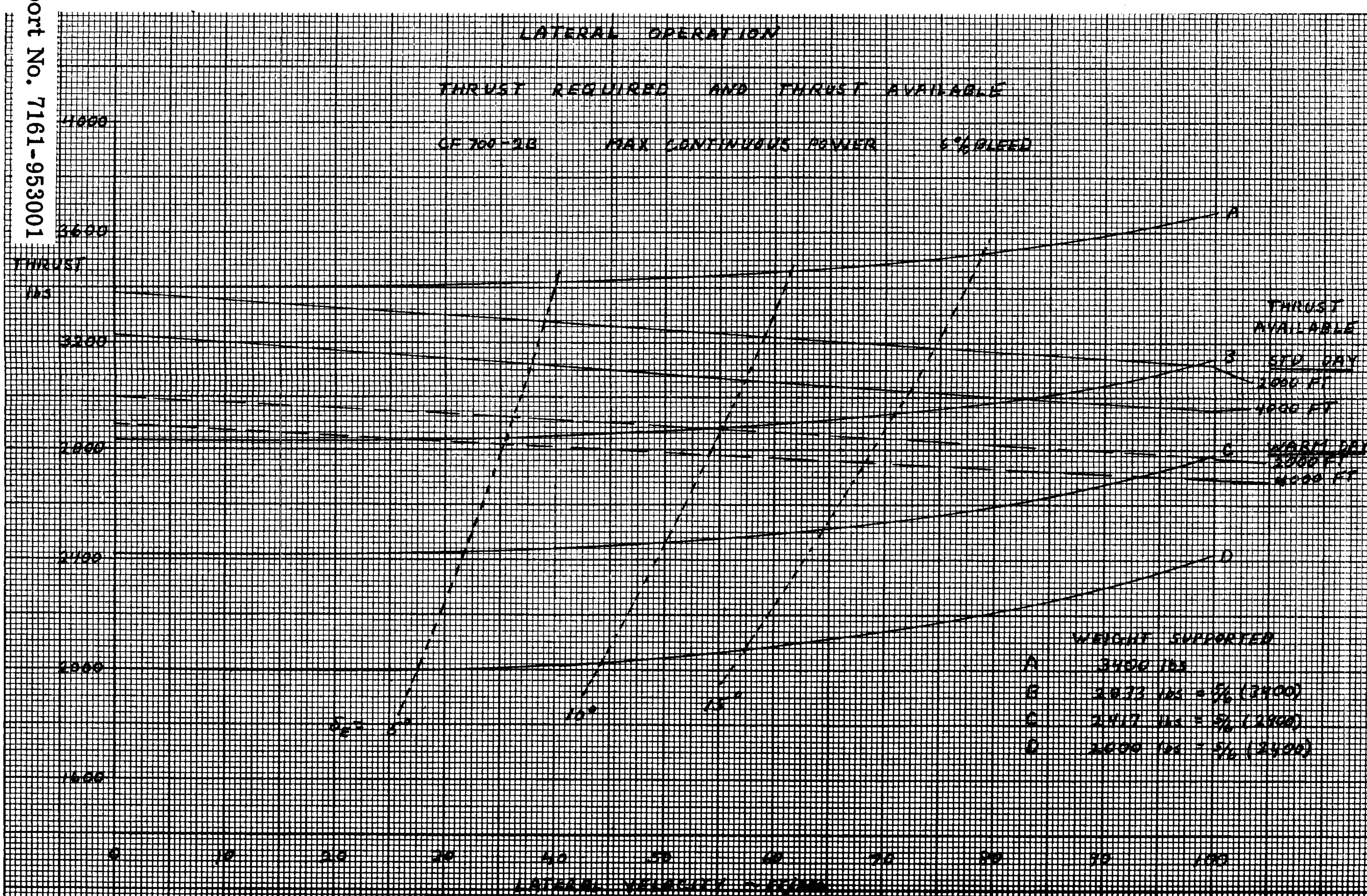


Figure V-6. Lateral Operation - Thrust Required and Thrust Available

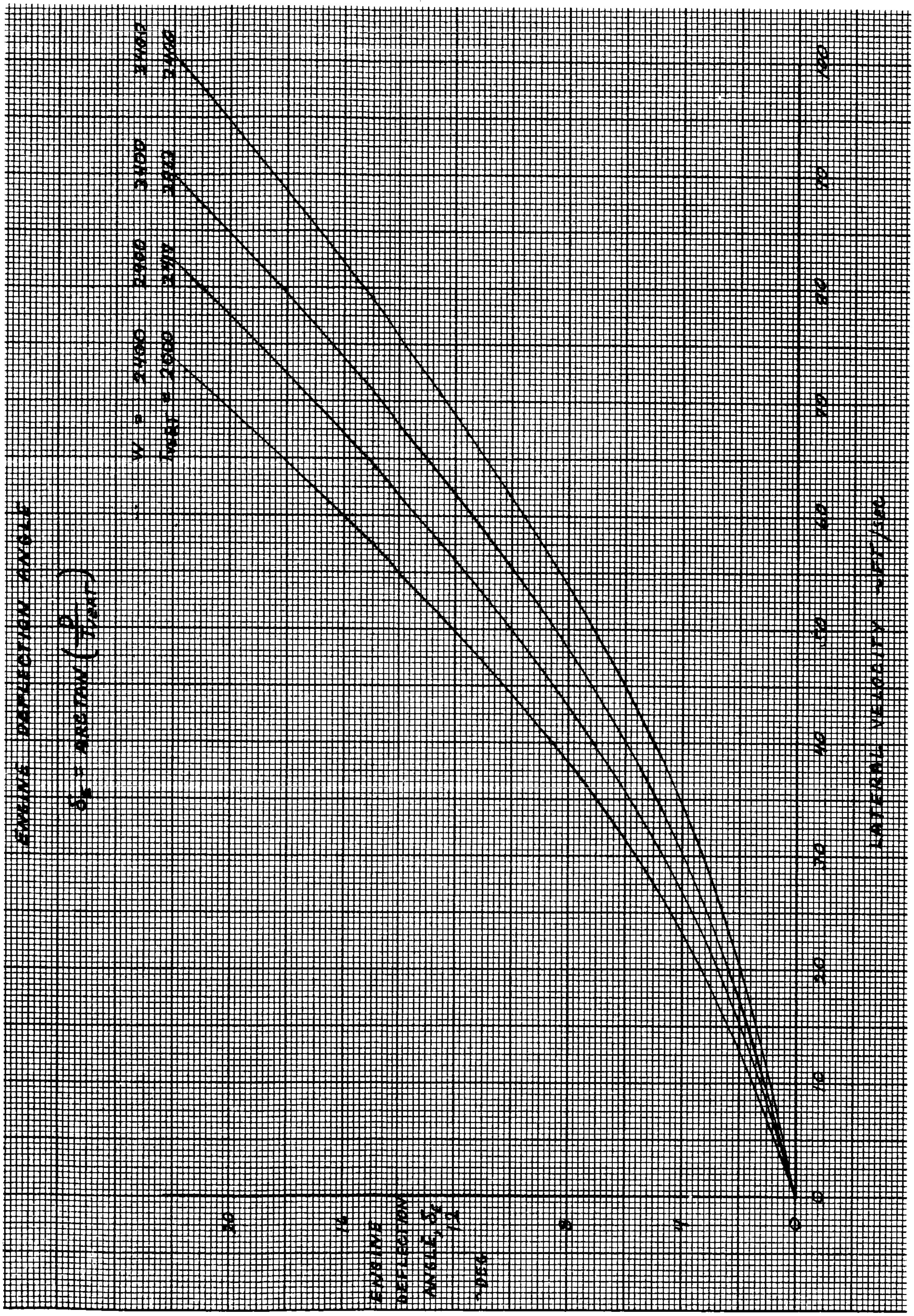


Figure V-7. Engine Deflection Angle

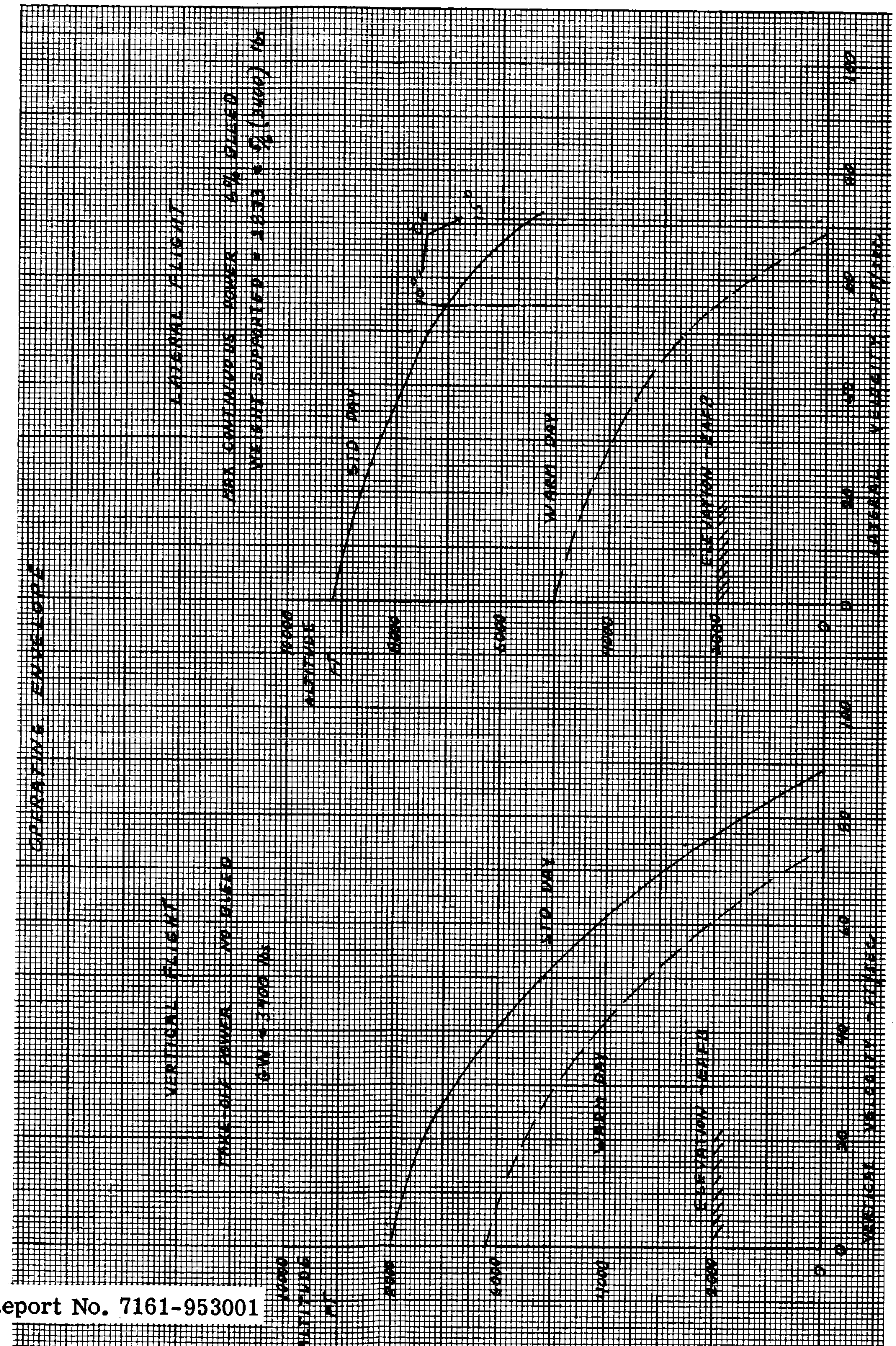


Figure V-8. Operating Envelope

3. Descent

In normal operation, the vehicle will descend from its initial altitude to a position on or near the ground. During this mode of flight, it is necessary only to reduce the engine throttle setting to adjust for the drag force, which in descending flight, is tending to support the vehicle. The only factor of importance is to initiate deceleration in sufficient time to avoid a hard impact. Normally, since a lunar landing is being simulated, the vehicle will be brought to a hovering condition with the lift rockets, while the jet engine throttle is automatically controlled to maintain thrust equal to five-sixths weight.

4. Emergency Descent

There are four possible types of power loss which may occur during a flight. These are in order of seriousness:

a. Failure of the jet engine.
b. Failure of the vehicle attitude control system.
c. Failure of the lift rocket system
d. Failure of the bleed air reaction system.

The first, failure of the jet engine, will cause loss of the vehicle. The other three, under most flight conditions, can be compensated for and the vehicle can be brought to a safe landing. The pilot needs only to cage the jet engine gimbal and take over manual control of the jet throttle. Stabilization and control will be by the vehicle reaction controls which are redundant. After a short flight interval, when gross weight has decreased, the engine bleed air reaction system can be used to stabilize the vehicle. This method cannot be used early in flight because in using the bleed air, the available jet thrust is reduced below takeoff weight.

The recovery envelopes are shown in Figure V-9 as a function of vertical descent velocity. A failure during descent is the most critical, since the vehicle will lose more altitude by the time it is brought to a stop than it would during ascending or lateral flight.

Although the vehicle at full gross weight cannot be recovered on a warm day if failure (b) occurs, it can be brought to a safe landing if the weight is less than about 3200 lbs, which should occur duringthe first 20% period of the flight.

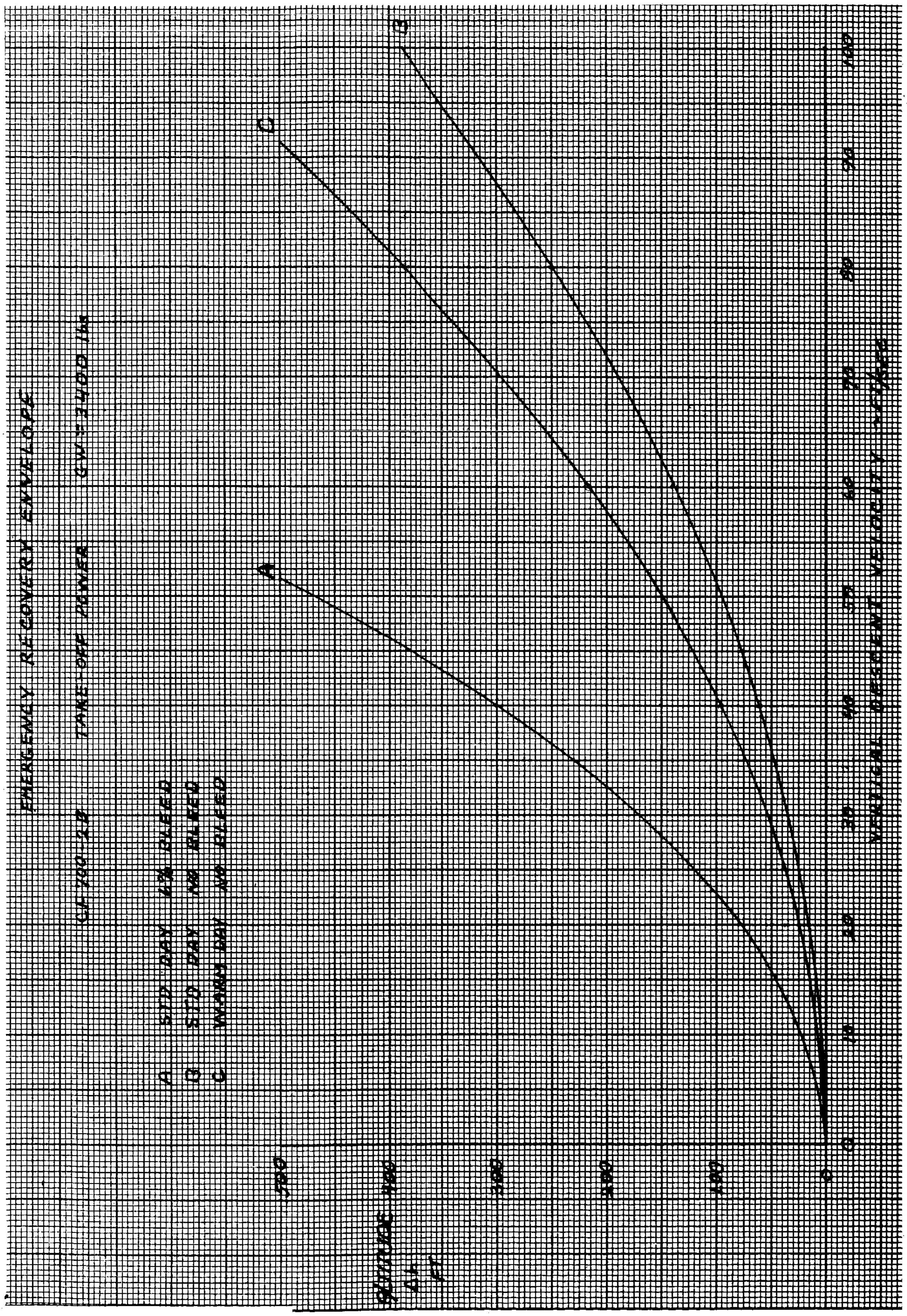


Figure V-9. Emergency Recovery Envelope

TABLE V-1

TAKEOFF AND CLIMB FROM 2000 FT TO 4000 FT

GW = 3400 lbs	Takeoff Power		No Bleed
	(R/C) AVG ft/sec	Time - Min	Fuel Consumed - lbs
Std. Day	70	.48	21
	50	.67	28
	30	1.11	43
Warm Day	50	.67	28
	30	1.11	44

TABLE V-2

ENDURANCE

	Velocity ft/sec	Fuel Consumption lbs/hr	Flight Time for 300 lbs of Fuel min.
Std. Day	0	1670	10.8
	50	1770	10.2
Warm Day	0	1830	9.8
	50	1920	9.4

VI. ROCKET SYSTEM

A. SYSTEM REQUIREMENTS

The rocket systems on a free flight lunar landing simulator must provide a good simulation of the lift rockets and attitude control rockets which would be employed on an actual lunar landing vehicle. The rocket system proposed for this vehicle has been designed to the following basic requirements:

(1) The simulator system must approximate the speed of response of rocket engines which will be used on a lunar vehicle for lift and attitude control.

(2) Since the exact configuration of an actual lunar vehicle has not been established at the present time, the simulator system must be flexible to simulate a variety of control systems.

(3) To provide an early low cost vehicle, maximum use must be made of existing components and systems.

(4) Since the simulator is manned, systems must be man-rated or have a background of experience to assure safe operation.

(5) Since attitude control of the vehicle is mandatory for safe landing, the attitude system must have high reliability.

(6) The system should utilize available propellants with which field personnel have had experience.

(7) The system should be of minimum weight consistent with performance requirements.

Experience with other VTOL vehicles, and analog simulation studies on this vehicle, indicates maximum control accelerations required in all axes to be less than one radian/second^2, for a flight duration of ten minutes. It is required that the control torques be proportionally controllable by the pilot and electrically controllable by a rate feedback system to provide attitude damping.

The rocket lift system for the lunar landing simulator must provide thrust of approximately 1/6 vehicle weight. Since the jet engine has sufficient thrust to fly the simulator without utilizing the lift rockets, no redundancy need be supplied in the lift rocket system. For a vehicle gross weight of 3500 pounds, 1/6-g results in a lift rocket thrust requirement of 580 pounds maximum. In addition, the lift rockets must provide this same vertical thrust component while the vehicle is tipped at a 30° angle relative to earth for horizontal translation.

$$\frac{\text{Thrust}}{\cos\theta} = \frac{580}{.866} = 670 \text{ pounds thrust}$$

Also, some capability should be provided for pilot control of thrust above and below the nominal. As a result, 1000 pound thrust was selected as the maximum required.

The lift rocket system and the attitude system are combined as a means of reducing complexity, weight and cost. The combined requirements are shown in Table VI-1.

TABLE VI-1

ROCKET IMPULSE AND THRUST REQUIREMENTS

10 minutes flight time for attitude controls.

2 minutes flight time for lift rockets.

Complete redundancy required for attitude control.

Attitude changes to be achieved without cross coupling or translation.

Lift System Total Impulse 500 pound average thrust x 120 seconds	= 60,000 # Sec
Attitude Control System Total Impulse 5 pound/minute (from analog simulation study) 100 sec avg I_{sp} x 50 lb = 5000 # Sec x margin of 2	= 10,000 # Sec
Total Impulse Required (Minimum)	= 70,000 # Sec
Excess Tankage 35% (Payload tradeoff capability)	= 25,000 # Sec
Thrust Levels - Throttleable Inflight Attitude control (each nozzle) Lift System (total for two nozzles)	 = 8 # to 80 # = 200 # to 1000 #

B. DESCRIPTION

Two basic rocket systems are available which can meet the requirements. One system uses hydrogen-peroxide as a monopropellant and the other uses nitrogen-tetroxide and hydrazine. Characteristics of both systems are shown in Table VI-2, calculated for the same total system impulse. The bipropellant system has a significant weight advantage and operational advantages of increased temperature range and loaded storage time. Components for a bipropellant system are under development at the present time. On the other hand, man-rated peroxide components are available from the X-15, and Mercury programs. In addition, much field experience has been gained on peroxide systems with the X-1, the X-15, and Mercury programs. Also, peroxide servicing equipment and experienced personnel are available at the NASA-Edwards Flight Research Center. Because of the early availability and low cost desired for the vehicle, the peroxide system has been selected. However, a discussion of the bipropellant system is included elsewhere in this report because it is considered a good choice for a future vehicle.

TABLE VI-2

	H_2O_2	N_2O_4/50-50 Blend
System Weight (dry) (wet)	226 lbs 1056 lbs	245 lbs 660 lbs
Total Impulse	95,000 lbs sec	95,000 lbs sec
Temperature Range	40°F to 100°F	20°F to 120°F
Specific Impulse	122 secs	244 secs
Practical Propellant Loaded time limit	1 day	2 weeks
Price per pound of propellant	55.5 cents	32 cents
Response	60 ms.	20 ms.

A system has been selected which can be used in a variety of ways to simulate a broad spectrum of control methods. Figure VI-1 is a schematic of the system. The system provides attitude control in pitch, roll, and yaw with pure couples; that is, no translation accompanies a change in attitude.

The need for providing torque without translation requires four nozzles for each plane of control (i.e., two roll left and two roll right). Therefore, twelve units total are required. The yaw units, however, are not located in the plane of the vehicle centroid. Hence, four additional nozzles are required - to prevent cross couples when only one of the two yaw nozzles is operable. This effect is discussed to a greater degree in the section of flight safety.

The attitude system is split into two identical subsystems each containing eight nozzles. Either subsystem can be used alone although translation would accompany a change in attitude, thus requiring a greater degree of pilot skill. Similarly, a variety of attitude rates of change can be simulated by operating, pulse mode, with varying on to off times in the three axes.

The nozzles are controlled by two sets of valves, one proportional controlled by the pilot through direct mechanical linkage, and one electrical, controllable by the pilot or by an autopilot. These systems may be used separately on in parallel.

Functionally, three major subsystems are included.

Pressurization System

The nitrogen system, completely redundant for attitude system capacity, consists of two 16.5 I.D. titanium spheres, full flow five micron filters, manual operated isolation valves, and commercial type ground adjustable pressure regulators. The nitrogen spheres are located diametrically opposed on the vehicle so that only the nitrogen lines and the two isolation valves are routed above the pilot compartment floor. As in the Mercury man-in-space capsule, the pilot has direct control over the high pressure stored gas and the peroxide tanks can be pressurized slowly to minimize impact stresses on the propellant feed system.

The nitrogen tanks will be charged to a maximum of 3000 psia at 100°F and are designed with a safety factor of 2 for man carrying application.

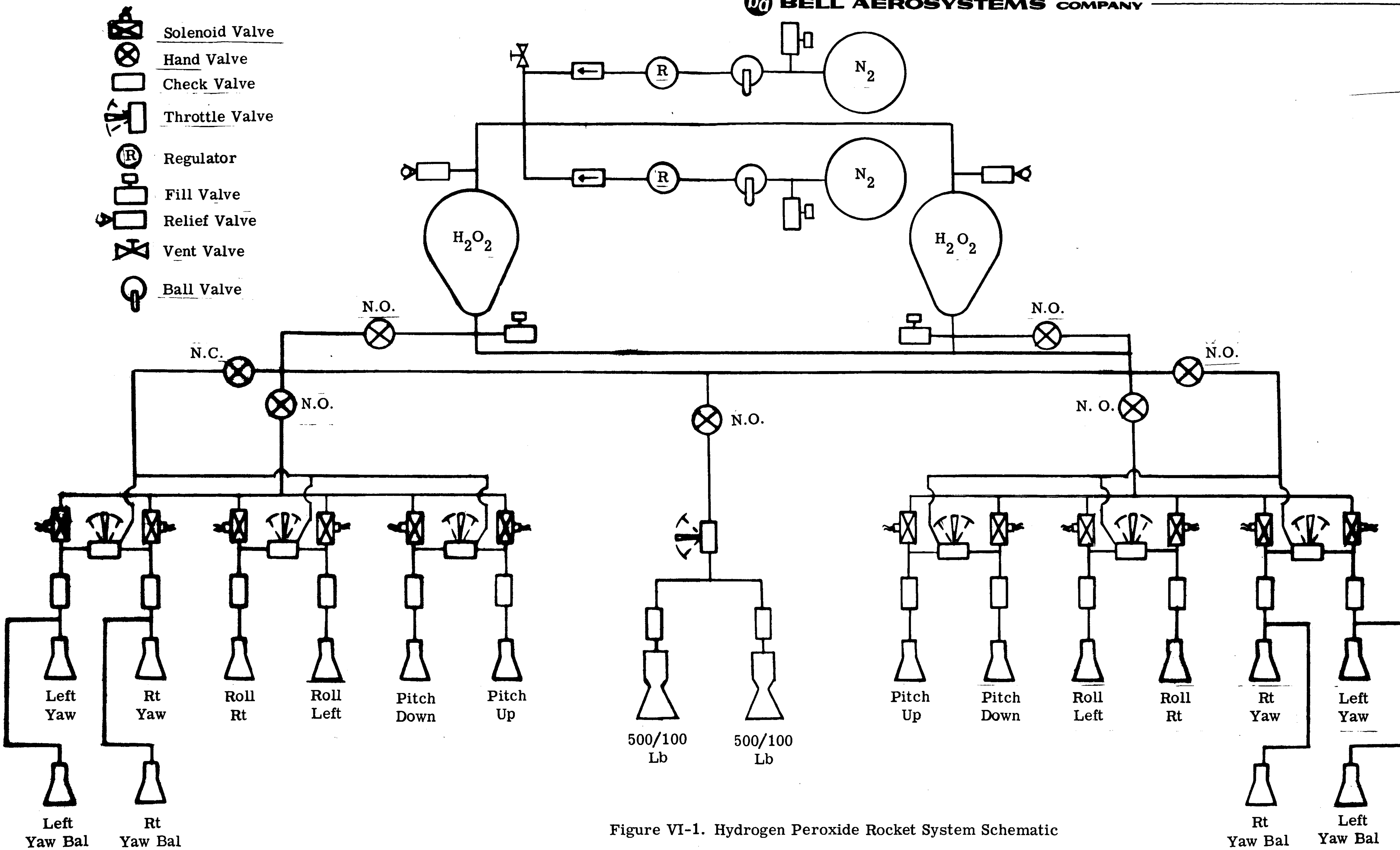


Figure VI-1. Hydrogen Peroxide Rocket System Schematic

Propellant Supply System

The propellant supply system has a capacity for 800 pounds of usable hydrogen peroxide located in two spheres on the vehicle struts fore and aft of the pilot's platform. Each sphere is approximately 25" I.D., and is constructed of 6061 aluminum alloy. Details of the tank construction are shown on the cross-section sketch, Figure VI-2. Two special considerations in the design selected were occasioned by the need to tip the vehicle axis for translation. One aspect resulting is the flow of propellant from the high tank to the low during translation, causing vehicle unbalance. If one minute of translation at the maximum pitch angle of 30° is used as criteria, all the propellant would shift to the lower tank. To minimize this problem, orifices will be located between the tanks such that the differential consumption, or transfer, would be limited to 2 pounds/second of flight.

Secondly, tipping the vehicle during the end of flight would mean the last 10 percent of propellant would be unusable if a spherical tank with a bottom outlet fitting is used. The sump or outlet cone shown on Figure VI-2 reduces the trapped prpellant to a negligible value. Cruciform aluminum baffles are installed in this sump region of the tank to minimize slosh and vortex problems. Each tank will be fabricated along conventional lines using automatic welding equipment to ensure high strength homogenous joints. Subsequent to assembly, heat treat, and pressure test, each tank will be thoroughly cleaned, anodized and conditioned in hydrogen peroxide. The tank is fundamentally similar to the Project Centaur tank which is approximately 22 inches in diameter, constructed of two 6061 spum aluminum shells and mounted on trunnions in the same fashion.

Either of the two peroxide spheres can be isolated and control of the vehicle maintained using the propellants in the other tank. Valves used for isolation are normally open, push-pull valves operated by the pilot. This is the method used by the Mercury capsule pilot.

Nozzle System

Sixteen thrust units firing in pairs comprise the attitude system. These have 3/8" solenoid valves of the type used on the Centaur program with check valves downstream of the solenoids to permit incorporation of manual throttle valves as an alternate mode of operation. Manual push pull valves will permit isolating either bank of eight units in the event of a valve failure. The lift system units have manual throttling valves similar to the valve used for the Bell Aerosystems Company rocket belt.

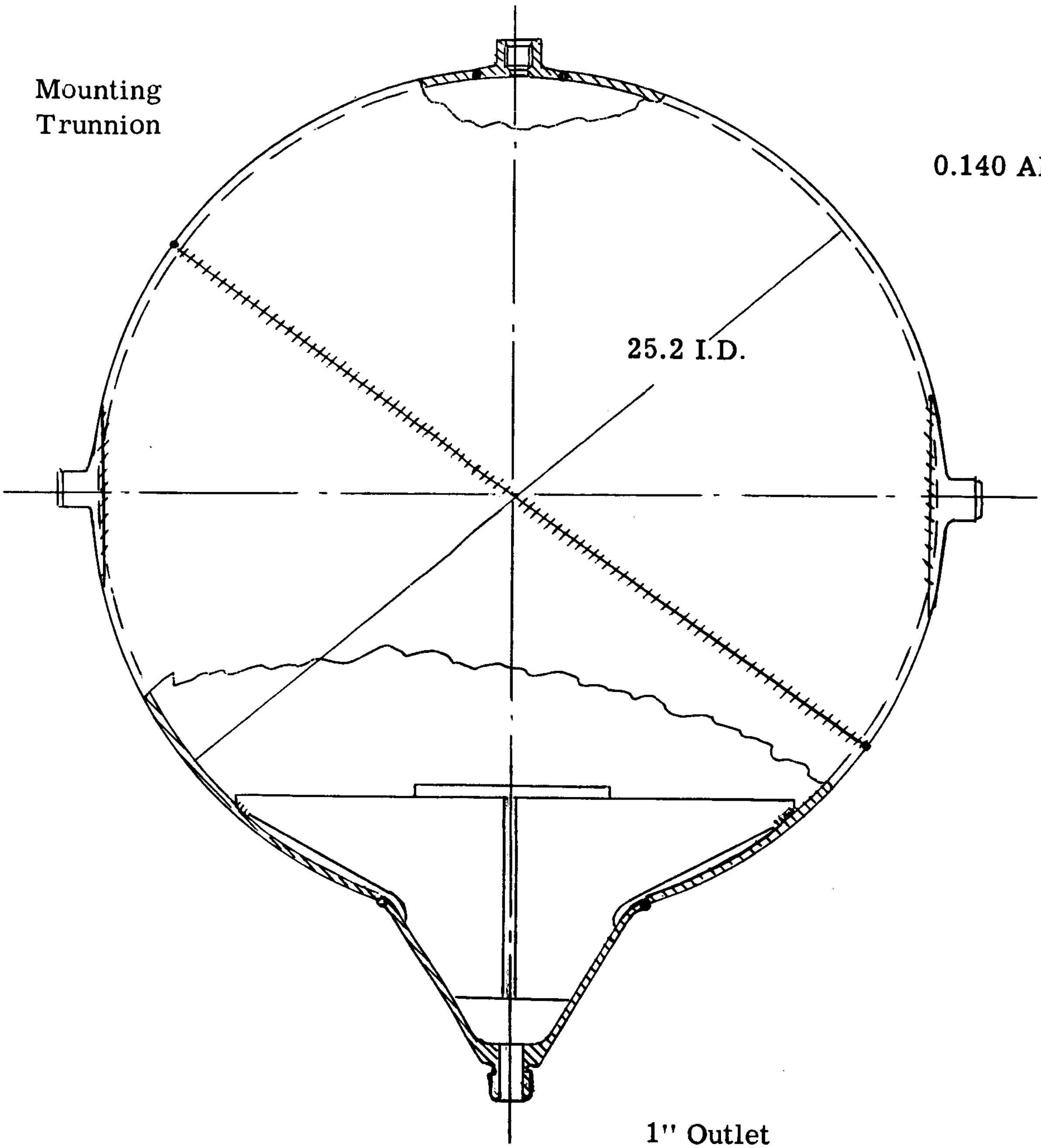


Figure VI-2. Hydrogen Peroxide Tank

The monopropellant thrust chambers for attitude and lift use 90% hydrogen peroxide and are adaptations of fully developed designs proven in many similar applications (X-15, Project Centaur, Flying Belt). The thrust chamber can be used with proportional throttling or in a fixed thrust pulse mode operation. The design of the proposed thrust units is shown in Figures VI-3 and VI-4 and can be accomplished as a routine process in a minimum of time. No analytical or development test effort is required. The fabrication of the thrust chamber hardware is straightforward. Fabrication of the only critical component, the catalyst bed, is controlled by a fully developed Bell process specification which guarantees performance and quality.

For the man rated flight application, a minimum qualification test program is recommended.

(a) Demonstration of thrust output and specific impulse

(b) Pulse characteristics and stability

(c) Environmental performance at +120°F

(d) Vibration and shock

(e) Endurance and thrust degradation

System weight is shown in Table VI-3.

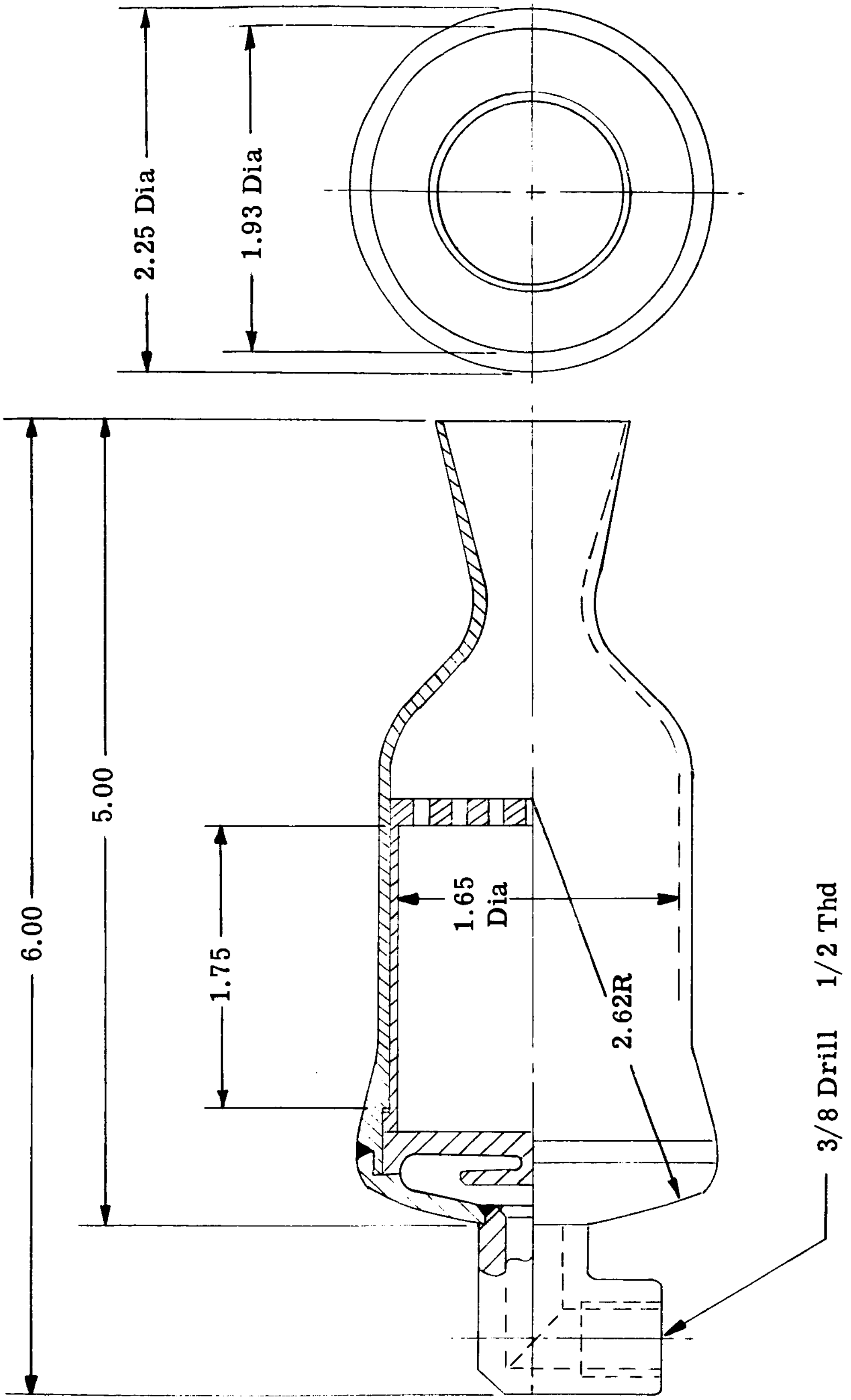


Figure VI-3. 80-Pound Thrust Chamber

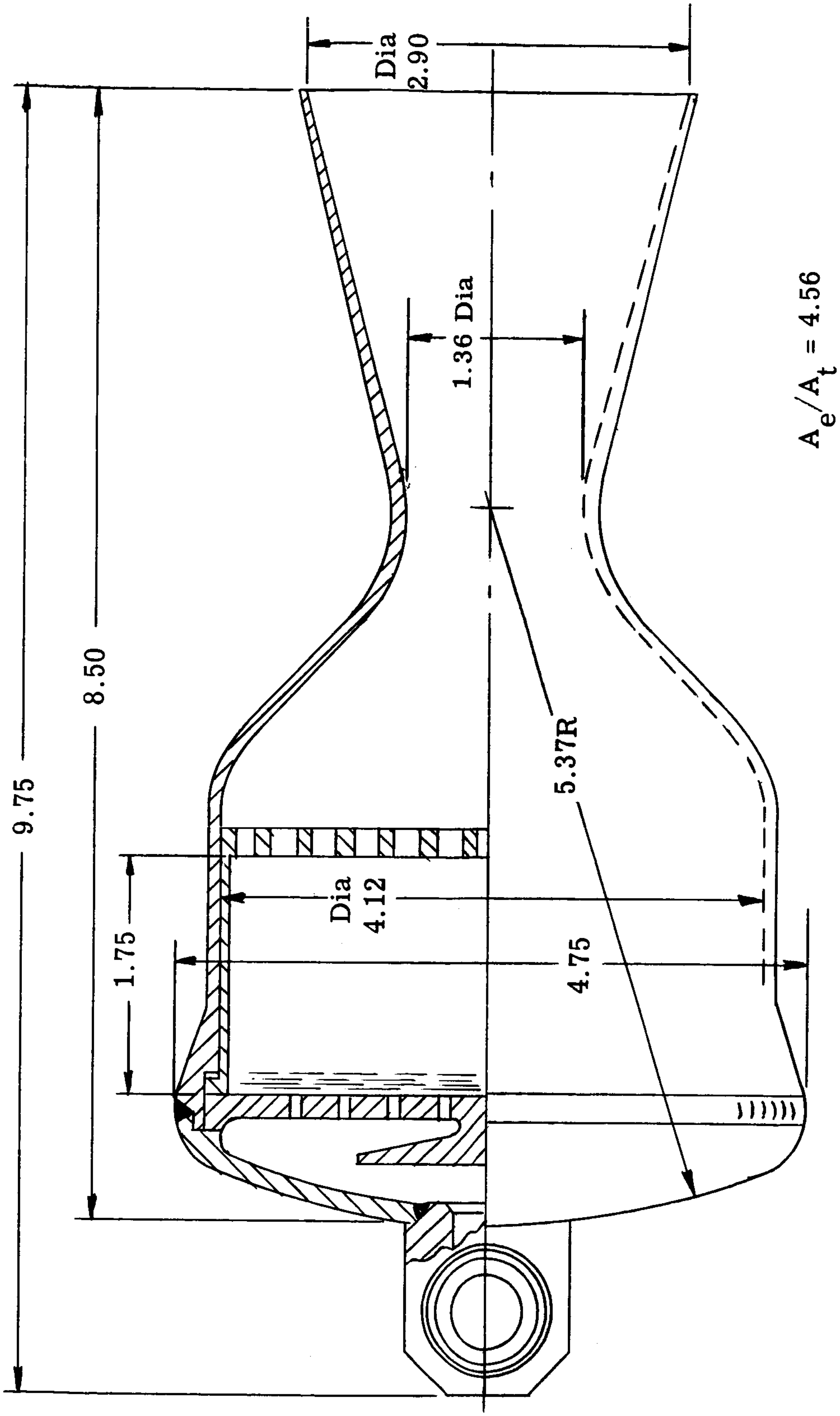


Figure VI-4. 500-Pound Thrust Chamber

TABLE VI-3

SYSTEM WEIGHT

	Qty	WEIGHT Each	WEIGHT Total
Nitrogen Spheres	2	25	50
N_2 Fill	2	.1	.2
N_2 Filter	2	.5	1.0
N_2 Isolation Valves	2	.5	1.0
N_2 Regulator	2	1.1	2.2
Relief Valve	2	.5	1.0
Tank Vent	1	.1	.1
Tank Fill	1	.3	.3
H_2O_2	2	43.5	87.0
H_2O_2 Tank Isolation Valves	2	.8	1.6
500 Lb Thrust Throttling Valves	1	1.6	1.6
500 Lb Thrust Chambers	2	6.8	13.6
500 Lb Thrust Check Valves	2	.1	.2
500 Lb Thrust Isolation Valves	1	.7	.7
80 Lb Thrust Throttling Valves	6	1.3	7.8
80 Lb Thrust Chambers	16	2.4	38.8
80 Lb Thrust Check Valves	16	.1	1.6
80 Lb Thrust Solenoid Valves	16	.3	4.8
80 Lb Thrust Isolation Valves	4	.5	2.0
Plumbing Brackets, etc.			10.0
Dry Weight Total			225.5
H_2O_2			600.0
N_2			27.0
Gross Weight Total			852.5

C. PERFORMANCE

System performance is dictated by the decomposition chamber efficiency, chamber pressure, nozzle configuration and temperature of the catalyst bed. Figure VI-5 shows the envelope of anticipated start delays for the first pulse. If the off time between pulses is less than one minute the chamber will remain warm and steady state specific impulse can be assumed.

Table VI-4 presents the performance characteristics of the two types of thrust chambers.

TABLE VI-4

80 Lb Thrust Chamber	
Thrust (Nominal)	80 Lbs
Throttle Range	80 to 8 Lbs
Specific Impulse (Sea Level Min.)	122 Sec
Chamber Pressure (Nom.)	250 psia
Propellant Feed Pressure (Nom.)	400 psia
Propellant Flow Rate (Max)	0.656 Lb/Sec
Start Response (1st Start) (Propellant Temp. +60°, +120°F)	150 Milliseconds
Start Response (Hot Catalyst Bed)	40 Milliseconds
Catalyst Bed Service Life	2 hours
Weight	2.4 Lbs
500 Lb Thrust Chamber	
Thrust (Nominal)	500 Lbs
Throttle Range	500 to 100 Lbs
Specific Impulse (Sea Level Min.)	122 Sec
Chamber Pressure (Nom.)	250 psia
Propellant Feed Pressure (Nom.)	400 psia
Propellant Flow Rate (Max.)	4.1 Lb/Sec
Start Response (1st Start) (Propellant Temp. +60°F, +120°F)	200 Milliseconds
Start Response (Hot Catalyst Bed) (Signal to 90% Thrust)	60 Milliseconds
Catalyst Bed Service Life (5% Thrust Degradation)	2 Hours
Weight	6.8 Lbs

Figure VI-6 shows the theoretical and actual relationship between thrust coefficient and chamber pressure for the constant nozzle area ratio of 4.35 used on the X-1A 75 pound thrust noll control rockets.

Figure VI-7 shows actual and theoretical C* versus chamber pressure for the X-15 112 pound thrust unit which is similar in size to the attitude control units.

D. SAFETY ANALYSIS

Safety for the hydrogen peroxide propulsion is extremely important and must be given paramount consideration in the design of any new system. Fortunately, there is a great deal of data and experience available so that the system will be inherently reliable. Perhaps more important is the adequacy of in process and field procedures to control cleanliness. Impurities in the hydrogen peroxide, storage at elevated temperatures, improper purging, all contribute to short service life and field failures.

The system will be designed so that failure of any component will not jeopardize pilot safety. Relief valves are incorporated on each hydrogen peroxide tank to permit isolation of either system and vent the tanks if a regulator malfunctions. Relief valve capacities are selected to maintain propellant tank pressure levels at the working pressure level for worst case conditions - regulator stuck wide open or regulator-seat extruded out.

In addition the pilot can manually isolate the high pressure stored gas. Gross leakage of propellant can be handled in a similar fashion - first, closing the ball valve and shutting off the source gas supply to the peroxide tank, then closing the tank isolation valve.

If it becomes necessary to isolate either tank, the remaining one would supply propellants to the attitude control system and the jet engine would be used to supply all lift required to return to earth. Sufficient nitrogen is available in one tank for attitude control during the descent to earth. Should any of the attitude control solenoids fail to open, its opposite unit would still function normally although some translation would accompany a change in attitude. Failure of a solenoid or check valve to close after an impulse bit would necessitate manual shutoff of that half of the system. Similarly, if the manual valve failed to return to neutral, the manual system can be isolated and operation continued, using the electrical system.

Many years of experience indicates that hydrogen peroxide exhaust is not a hazard to personnel or equipment. Peroxide exhaust consists of water vapor and free oxygen at relatively low

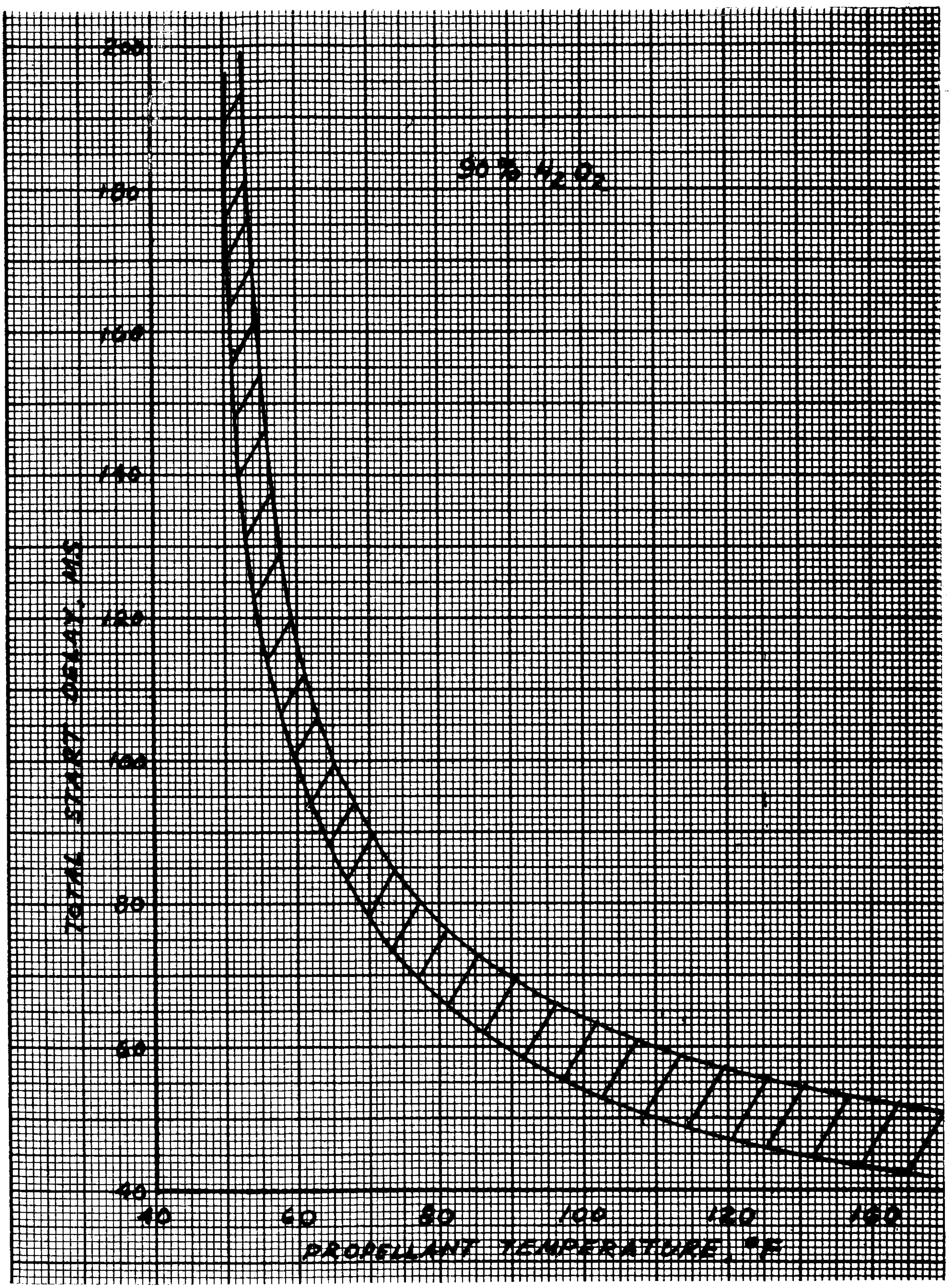


Figure VI-5. Total Start Delay (1st Pulse) versus Propellant Feed Temperature

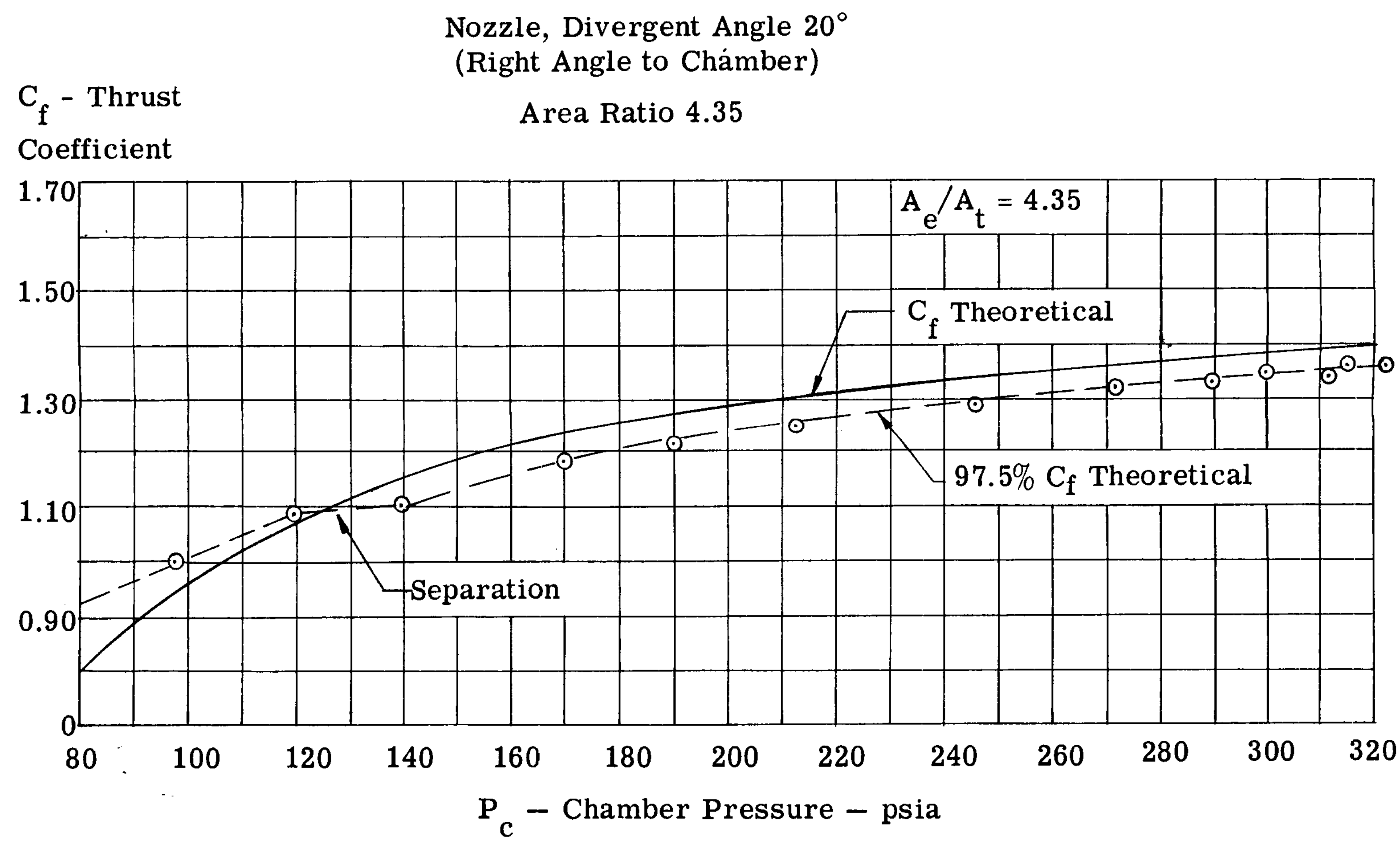


Figure VI-6. X-1A Roll Control Thrust Chamber 75-Pound Throttleable Model 130-470-010-1 S/N 4

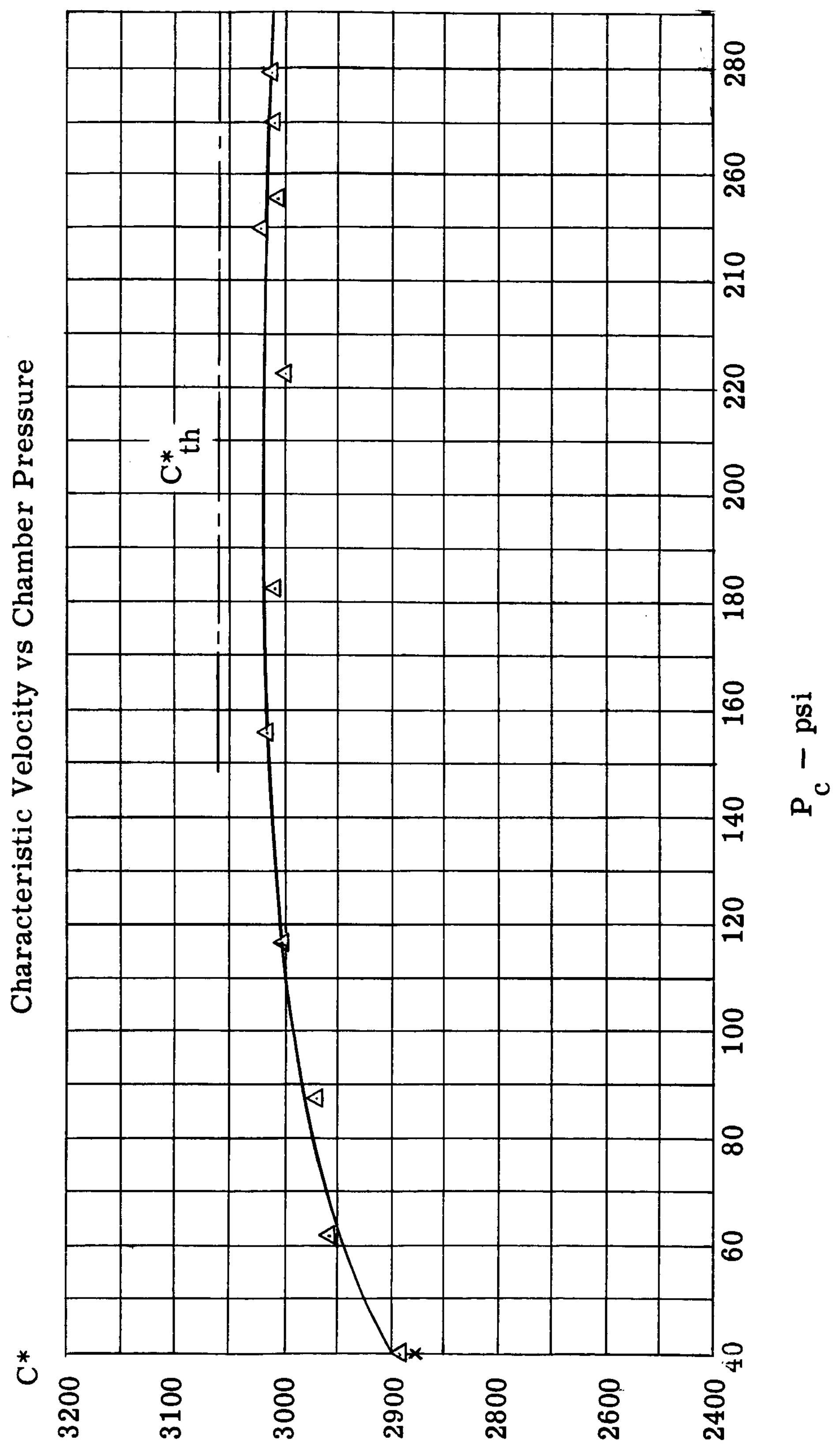


Figure VI-7. X-15 Yaw and Pitch Thrust Chamber 112-Pound Throttleable Model 130-470-021 temperature. Experience with the Bell Rocket Lift Belt, in which peroxide nozzles are located close to the man's shoulders, indicate no danger from exhaust impingement on the clothed body. In this vehicle, the peroxide exhaust may mix with the jet engine exhaust. This should not be a problem even if the jet engine is run fuel rich. The jet engine aft fan exhausts 100 pounds of air per second which contains 20 pounds of oxygen. The lift rockets running at full throttle will add only three additional pounds of oxygen per second.

E. BIPROPELLANT SYSTEM

During the propulsion system study, system weight considerations led to investigating a liquid bipropellant system. For comparison, the same total impulse and thrust levels were assumed as were utilized for the selected hydrogen peroxide system. The equivalent system is shown schematically in Figure VI-8. As can be seen, the same functional redundancy is incorporated. Isolation valves are intertied mechanically to minimize operational complexity and to maintain a high degree of system reliability. The bipropellant system will permit longer storage times, without degradation in specific impulse, and the smaller propellant volumes and transfer lines will minimize propellant center of gravity shifts. Another advantage of a bipropellant system is the inherently wider operating temperature range. Perhaps the most significant advantages are the short response times and consistent thrust reproducibility over wide temperature ranges available with bipropellant systems.

Some early flight training with a bipropellant system will shorten the ultimate development time for the actual lunar landing vehicle which, undoubtedly, will utilize a bipropellant system.

Table VI-5 is a weight summary for the bipropellant system.

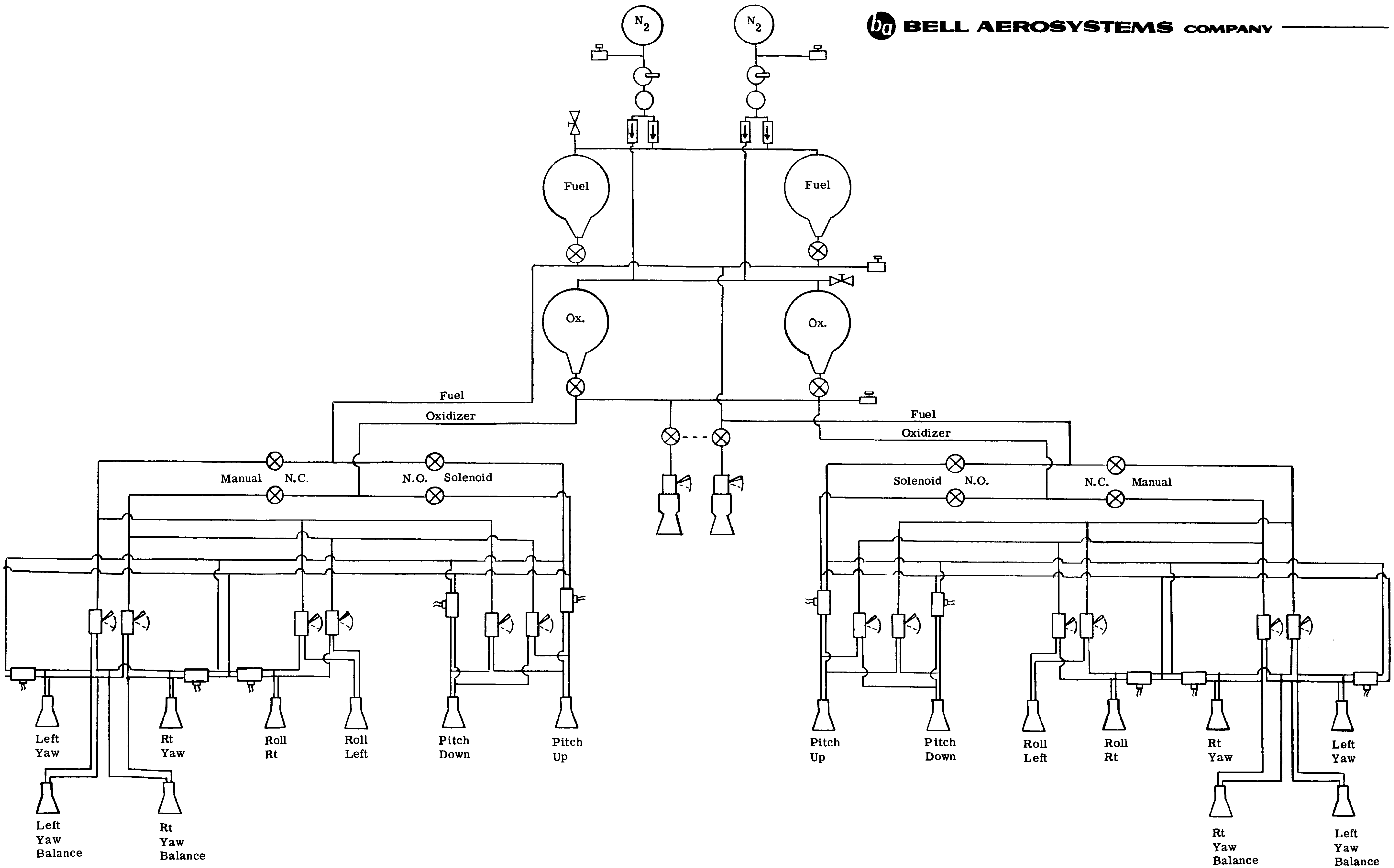


Figure VI-8. Bipropellant Rocket System Schematic

TABLE VI-5

BIPROPELLANT SYSTEM

	Qty	WEIGHT Each	Total
N_2 Spheres	2	12.0	26.0
N_2 Fill Valve	2	.1	.2
N_2 Filter	2	.4	.8
N_2 Isolation Valves	2	.5	1.0
N_2 Regulator	2	1.1	2.2
Relief Valve	4	.8	3.2
Vent Valve	2	.1	.2
Propellant Tank	4	7.0	28.0
Tank Isolation Valve	4	.8	3.2
500 Lb Thrust Throttle Valve	2	1.6	3.2
500 Lb Thrust Chamber	2	15.0	30.0
500 Lb Thrust Isolation Valve	2	.7	1.4
80 Lb Thrust Throttle Valve	12	1.3	15.6
80 Lb Thrust Chamber Assembly	16	7.0	112.0
80 Lb Thrust Isolation Valve	5	.5	2.5
Plumbing and Brackets			15.0
Dry Weight Total			244.5
Nitrogen			15.0
Propellants			400.0
Gross Weight Total			659.5

VII. COCKPIT

A. GENERAL

The cockpit on this vehicle consists of a semi-enclosed platform containing space for two side-by-side ejection seats. Flight control by one man is provided. The cockpit layout and equipment is provided for flight by VFR, except when instrumented lunar landing is being simulated. Although the function of this vehicle is to simulate a lunar vehicle, compromises have been made to provide flight safety on earth. In general, standard helicopter practice has been followed when this has been consistent with good lunar simulation.

B. ENCLOSURE

The pilot enclosure consists of a light tubular aluminum framework over which sheet Mylar is stretched. This protects the pilot from wind and provides a base on which to attach a colored coating to simulate limited window area. The pilot uses glasses of an opposite color. By removing the glasses he has full visibility. The enclosure is open at the top to provide clearance for seat ejection.

C. DISPLAYS

The minimum displays consistent with flight safety and hovering flight have been incorporated. These instruments are shown in Figure VII-1. VTOL experience has indicated the need for the basic flight instruments, altimeter, rate of climb indicator, and pitch and roll attitude indicator. Although air speed has no meaning in a true lunar vehicle, an air speed indicator is provided in this vehicle, to advise the pilot of his approach to limiting air speed conditions. A drift indicator has been incorporated because analog simulation results indicate that this is a valuable aid in hovering. Although jet engine instruments will not be required on a lunar vehicle, they are provided on this simulator as a convenience in ground engine runup and as a warning to the pilot of impending trouble which might initiate an emergency landing procedure. These include a jet fuel level indicator, engine rotor speed indicator, exhaust gas temperature indicator, lube oil pressure indicator, lube oil temperature indicator, jet thrust indicator, and fuel tank pressure indicator. Rocket display consists of a warning of low peroxide level. As a further convenience in ground and airborne check, a clock and A.C. and D.C. volt meters are provided. Warning lights are provided to indicate that the gimbal locks are locked, and that the jet engine has reached the maximum tilt angle allowed.

D. CONTROLS

Pilot control of vehicle attitude is accomplished with a conventional pitch and roll center stick and yaw pedals. These are provided for one seat and move with the seat when changing from a single seat to a dual seat configuration. This seat is also equipped with throttles controlling the jet engine and the lift rockets. The center stick, the pedals, and the throttles are mechanically connected to the systems which they control. Vehicle attitude is also controlled through a parallel electrical link. Attitude controls installed on the second seat will be electrically linked only. The attitude controls can be used in either a proportional or on-off mode, and can call for control acceleration, adjustable from one radian per second^2 down to .05 rad/sec^2. The rocket throttle gradient is adjustable with a detent calling for zero lift at one end and full throttle at the other end. The jet throttle control provides a manual override on the automatic throttle programmer.

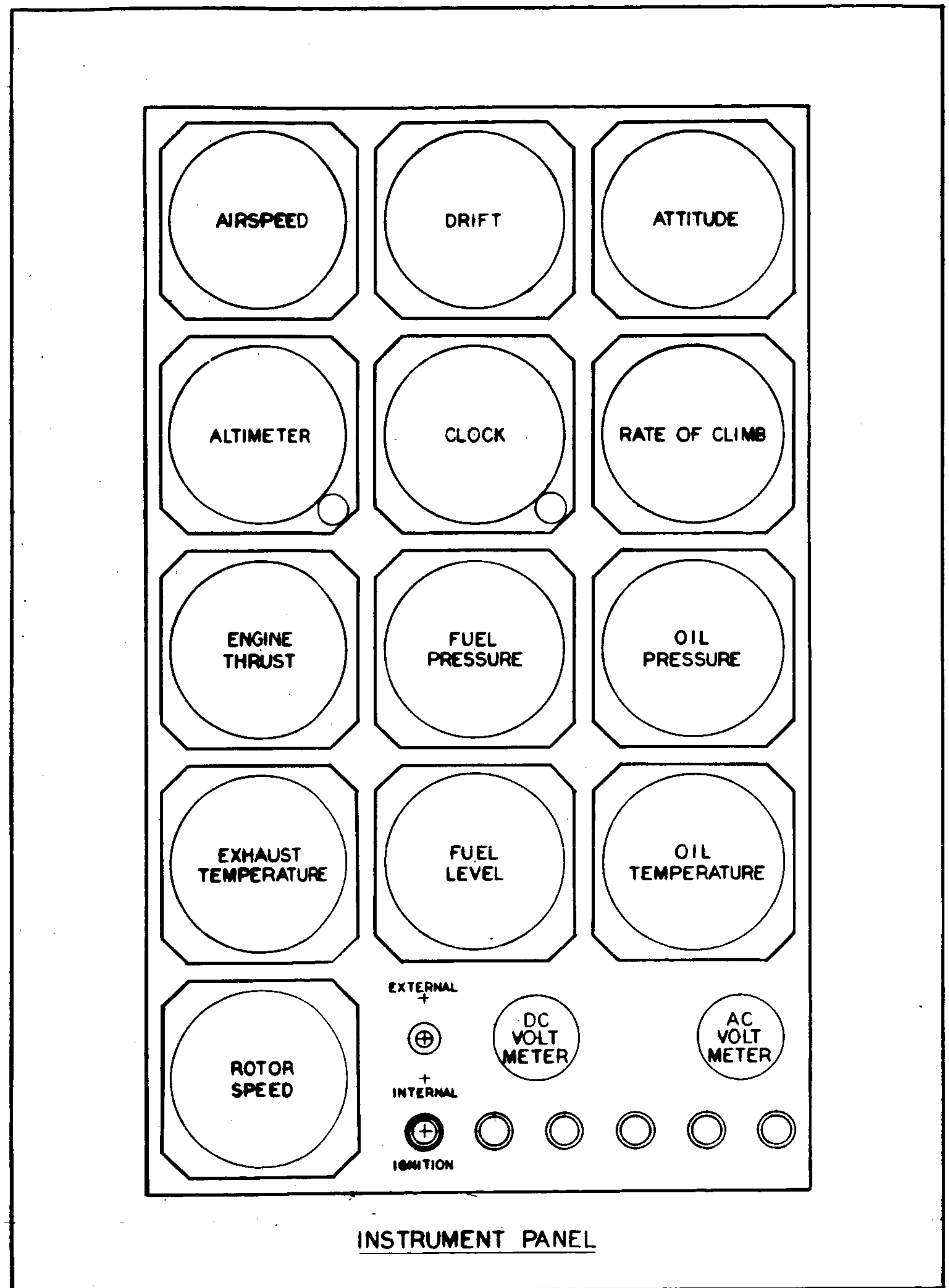


Figure VII-1. Instrument Panel

VIII. ELECTRICAL SYSTEM AND ENGINE STARTER

A. SYSTEM SELECTION

Three basic systems were investigated to determine their suitability as power systems for the lunar landing simulator. These are: (1) D.C. starter/generator system, (2) A.C. starter/generator system, and (3) air impingement starter/A.C. generator system.

The air impingement starter, A.C. generator system was selected because it is less costly than the A.C. starter/generator system, and provides a lower weight system than the D.C. starter/generator system.

1. D.C. Starter/Generator System

This system consists of a 28 volt D.C. - 200 amp D.C. starter motor/generator, a D.C./A.C. inverter, and associated controls and power distribution. The D.C. starter motor operates as a 28 volt D.C. generator when the engine is up to idle speed. A static D.C./A.C. inverter supplies 3 Ø 115 volt 400 ~ with a frequency regulation of ±1%. The weight breakdown for this system is:

starter generator G.E. 2CM63	41 pounds
D.C./A.C. inverter (3KVA)	95
control and distribution	15
	151 pounds

The greatest weight penalty is for the conversion of D.C. to A.C. (3KVA - static). If the vehicle payload could utilize 28 volt D.C., this inverter would be eliminated and the D.C. starter/generator system would be recommended as first choice.

The vehicle flight control system requires approximately 200VA of 400 ~ power, which can be supplied by a 10 pound static inverter.

2. A.C. Starter/Generator

An A.C. starting A.C. generator system was also considered. This included the induction start synchronous generator, and the induction start induction generator.

The induction start synchronous generator requires a squirrel cage winding on the rotor to develop a starting torque. In addition, the field winding must be protected from high induced voltages during start, by closing the field circuit through a resistor. These complexities, plus a sequenced voltage starting control, were weighed against the problems of an impingement start. Air impingement start was found to be much simpler to implement.

The induction generation has the advantage of providing good motor characteristics, since it operates as an induction motor, but has a dual disadvantage when running above synchronous speed (as a generator). These disadvantages include a requirement to take lagging current from the line and the incapability to supply lagging current to the utilization equipment. Thus, the induction generator system was discarded in favor of the synchronous generator.

Manufacturers contacted indicated that an A.C. starter/generator is not available as a standard item. Development cost would be considerably in excess of the price of a D.C. machine.

3. Air Impingement - A.C. Generator

This system consists of a turbine air impingement start duct, A.C. generator, 3 Ø rectifier, associated controls and power distribution. A ground air compressor supplies the air required for starting the engine (100 pounds/minute at 43 psia at 360°F).

The A.C. generator is an 8-pole synchronous generator which can supply 400 ∿ 3 Ø power at 6000 RPM. In order to obtain a qualified unit, a number of vendors were contacted. A 5KVA Leland AGE 41-2 generator with built-in regulator was selected since it is available as an "off the shelf" unit. Leland has produced these units for the French Dassault fighters. The line to neutral voltage is adjustable between 115 — 125 volt A.C. and the regulation is ±5 volt at 6000 RPM. It is expected that the nominal speed will be 6500 RPM, which corresponds to 434 ∿. This will be the accessory pad operating speed during most of the flight; however, speeds down to 6000 RPM (400 ∿) at the end of the flight, and speeds up to 71,000 (474 ∿) at liftoff will be encountered also. This variation offrequency (434 ±10%) will not adversely affect the operation of the basic vehicle systems and is expected to be acceptable for most flight research equipment. If closer frequency tolerance is required, an A.C./D.C. inverter is recommended. Frequency tolerance of ±1% or better is available in static inverters with ratings up to 3KVA. The 3 Ø rectifier selected is a rotary transformer solid state rectifier type as made by Westinghouse. This unit utilizes a 3-phase wound rotor induction motor which acts as a fan and transformer. The unit is capable of delivering 35A at 28 volt D.C. The 35A capability allows for possible addition of up to 500VA D.C./A.C. constant frequency (400∿ ±1%) conversion equipment, if required during the flight test program.

The 3 Ø rectifier is an "off the shelf" unit 4" diameter x 7.5" long, and weighs approximately 7 pounds.

The weight breakdown for this system is:

5KVA (generator regulator) Leland AGE 41-2	32 pounds
3 Ø rotary rectifier (Westinghouse)	7
Distribution and controls	15
	54 pounds

A schematic of the system is shown in Figure VIII-1. The 3 Ø power is connected to three separate busses through an Int-Ext relay. The three line busses provide flexibility in connecting individual line to neutral loads (115 to 120 volt A.C.). Int-Ext control allows complete checkout of the simulator equipment from an A.C. ground power source.

The 3 Ø transformer rectifier provides 28 volt D.C. with less than 15% ripple. The internal D.C. system is not switched, but is parallelled with the external D.C. power. This eliminates the problem of control relay chatter or sequence interruption.

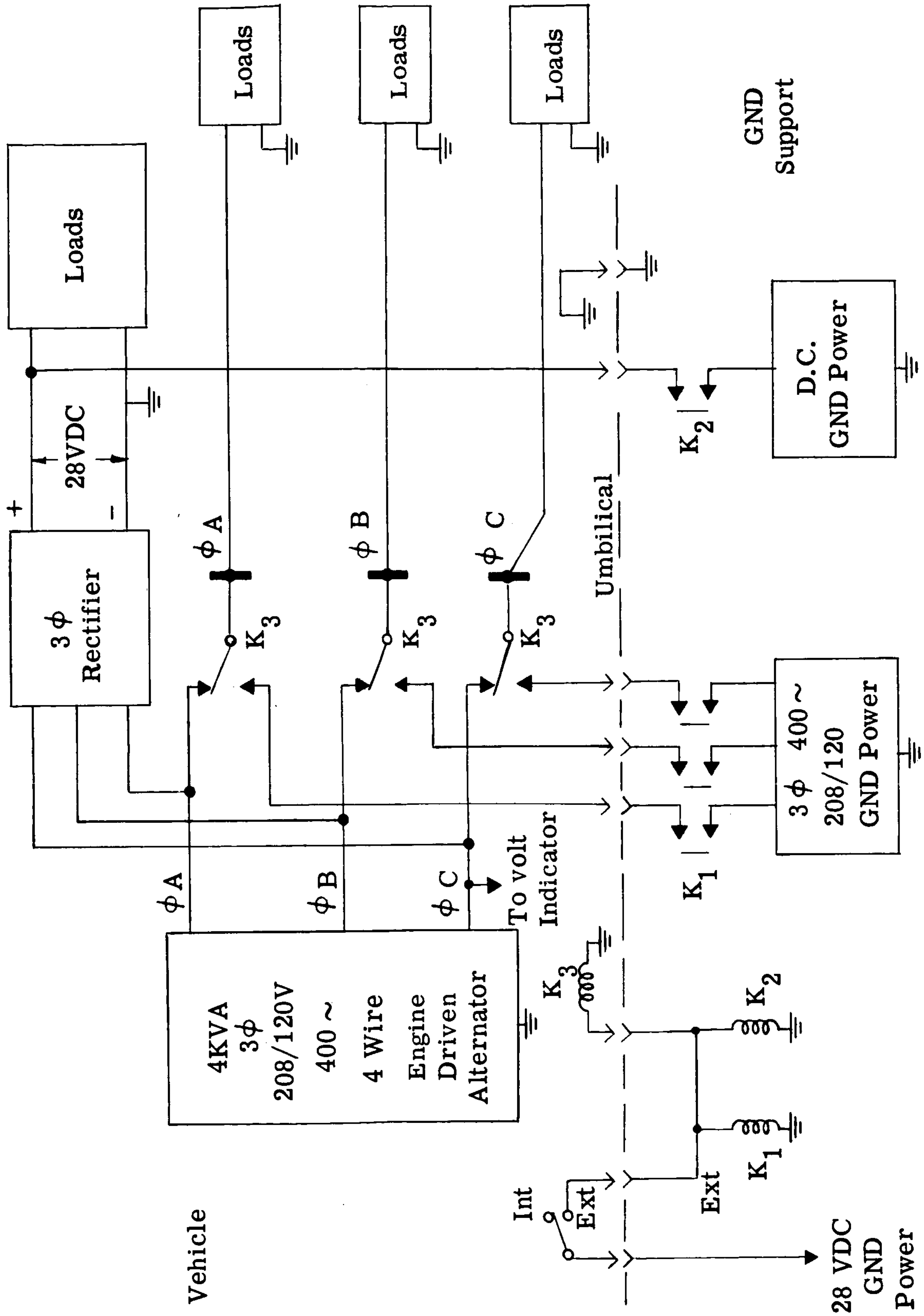


Figure VIII-1. Electrical Power and Distribution System

IX. RELIABILITY AND SAFETY

A. JET IMPINGEMENT EFFECTS

Three principal phenomena can be caused by ground proximity and the impingement of the ground of engine exhaust. The three effects are:

1. Ingestion into the engine of air mixed with exhaust products, leading to reduction of thrust.

2. Generation of aerodynamic forces between structure and air entrained by the deflected exhaust.

3. Erosion or damage to the ground surface with ensuing formation of debris.

Bell Aerosystems Company has conducted extensive tests on jet impingement phenomena. These tests include experiments with the first jet VTOL air test vehicle (now in the Smithsonian Institute, Washington), with the X-14 VTOL airplane, now operated by NASA, and with a 1/10-scale model of an eight-engine VTOL fighter. The latter model reproduces the effects of two J85 afterburning engines on each wing tip and of two pairs of non-afterburning J85 engines in the fuselage. The three previously listed phenomena were studied and the results are directly applicable to the problems of the Lunar Landing Simulator. More severe conditions were created by the engines of the scale test than are imposed by the Lunar Landing Vehicle's supporting jet.

Bell's experience showed that all the effects of jet impingement are influenced by the distance between exhaust nozzle and ground, by the quality of the ground surface, the temperature and kinetic energy of the jet at ground level, and by time. Steady state conditions may be expected to develop within seconds during hovering or slow displacement of the vehicle, with the possible exception of the formation of debris or of a dust cloud in cases that such could develop (see Section 3).

The exhaust of the CF700-2B aft turbofan engine used in the Lunar Landing Simulator consists of two parts: a hot inner jet and a much colder outer jet. The weight flow rate in the outer, cold jet is two times that of the primary airflow through the engine. The jet impingement phenomena are most pronounced at takeoff (or touchdown). The distance between ground and exit plane of the simulator's annular jet is four times the OD of this jet. At this distance the velocity in the composite jet decreases from an axial maximum of 1543 fps to 100 fps at a radius of 2 feet, the temperature from its axial maximum of 1190° to 200°F at 2 feet radius and to 150°F at a radius of 7 feet - all the values predicted by the engine manufacturer in an undeflected free jet. The intake of the engine is 6.65 annulus diameters above ground.

1. Ingestion of Air Mixed With Exhaust

Ingestion of exhaust-polluted air into the engine is not considered to cause any problem. This statement is based on a phase of Bell's experiments with the previously mentioned 1/10-scale model. In those experiments, the afterburning wing tip engines were operated alone and in vertical position. The fuselage engines were shut off. The configuration resembled that of the simulator. The operating engines were approximately 35 exit diameters apart, with their exhaust 2 diameters above ground. The temperature excess of the afterburning jets (at ground) was approximately 2900°F, the temperature increase of the intake air of the order of 10°F. The Lunar Landing Simulator's engine exhaust nowhere exceeds 1190°F. Consequently, the intake temperature will be increased by less than three degrees and the ingestion loss negligible, of the order of 25 pounds thrust, if the mean temperature excess of the fan engine's exhaust is assumed to be 500°F at 4 diameters and the intake temperature rise is set proportional to the temperature excess at impingement. The temperature coefficient of the thrust is approximately 10 lb/°F.

2. The Ground Effect

Since the air entrained by the engine's exhaust will have a low velocity and will meet very little resistance from the simulator's largely open framework structure, it is not expected to cause significant adverse or helpful ground effect.

The engine exhaust and induced flows are considerably decelerated by the time these arrive in the neighborhood of the reaction jets. The exit velocity of the latter is several orders higher than these. No significant deflection of the reaction control jets is anticipated.

3. Debris Formation

No debris will be formed by the Lunar Landing Simulator's engine-exhaust when operated above a hard surface. Bell operated both the VTOL air-test vehicle and the X-14 over conventional runway surfaces without causing any damage. The exhaust nozzles of these airplanes came as close to the ground as 1-1/2 exit diameters. The exhaust temperatures were comparable to that of the simulator engine's hot core. At 1-1/2 diameters from the exit, the mean dynamic head was 650 psf in one of the X-14's jets. At four annulus diameters away from the simulator's engine-exit, the dynamic heads are: 900 fps on the axis, 300 psf at a radius of 1 foot and 9 psf at a radius of 2 feet. The mean value within the area of 2 feet radius is of the order of 110 psf, well below that of the X-14's exhaust.

Conditions could change considerably if the simulator were to operate over unprotected ground. R. Kuhn, NASA TN D-56, describes experiments relating to erosion and debris formation due to jet impingement on surfaces ranging from dry sand to sod. The mean dynamic head at nozzle exit is his reference quantity. It is evident from this work that dry sand and loose dirt will be greatly disturbed by the simulator exhaust jet; only the order of 10 psf dynamic head was required to initiate disturbances on such surfaces. Wet sand or wet loose dirt also will be disturbed, but on a lesser scale, since about 100 psf mean head was required to creat a disturbance on these surfaces. On the other hand, wet or dry sod required the order of 1000 psf dynamic head to produce a disturbance; thus, operation over sod without significant debris formation appears feasible.

In summary, operation over conventional hardened surfaces and even over sod without significant surface damage or debris formation is feasible. Operation over unprotected loose soils can be expected to result in significant airborne debris formation and is not recommended.

B. RELIABILITY ANALYSIS

1. Objectives

The principal objective during the course of this study was to ensure that reliability and safety factors were appropriately considered along with other system factors, as trade-off studies were made, and design concepts and approaches were selected for the vehicle design. This will provide optimal assurance that in any follow-on development, design, production, and operational usage of the vehicle, adequate system reliability and crew safety can be provided at minimal cost.

2. Vehicle Reliability and Crew Safety

The reliability evaluation of the lunar landing simulator covers the following three major factors:

(a) Probability that the vehicle will accomplish the specified mission successfully.

(b) Probability that the vehicle will be returned safely to the ground, once committed to flight.

(c) Crew safety.

The preliminary reliability analysis of the proposed vehicle is based upon the present mission requirements, the design of the vehicle, and the operational requirements of each vehicle system obtained from the results of the current study. As the vehicle and/or mission requirements are definitized during the course of a follow-on vehicle fabrication and test program, the analysis can be modified, augmented, and refined.

For purposes of this analysis, the mission requirements of the lunar landing simulator are considered to consist of a total mission time of 10 minutes. This includes vehicle lift-off to specified altitude, simulated lunar maneuvers at altitude, and vehicle descent to ground.

The low estimated component failure rates for most of the vehicle systems are based upon experience with these systems in actual operational use, such as the Bell-produced reaction control systems for Mercury, Centaur, X-15, and Agena, and the General Electric J85 engine.

The preliminary reliability estimates of the proposed vehicle systems with component failure rates are presented in Table IX-1.

3. System Reliability Analysis

The probability that the vehicle will accomplish the specified mission requires that all vehicle systems (except crew safety provisions) must operate successfully.

The total failure rate of the vehicle systems operating in automatic, fly-by-wire, or manual mode for one 10 minute mission is shown in Table IX-1 and detailed in Table IX-2.

TABLE IX-1

	System	Estimated Failure Rate Failures/10^6 Missions
A	Main Propulsion and Engine Attitude Control*	6089
B	Reaction Control and Lift Rockets*	714
C	Electric Power	1400
D	Vehicle Structure	700
		8903

* The failure rate of the autopilot system is included in these systems as indicated in Table IX-2, Items A.1 and C.1. The reaction control system includes the redundant automatic and manual mode of operation.

Therefore, the estimated mission reliability of the vehicle is R = 99.11%.

TABLE IX-2

A. Main Propulsion and Engine Attitude Control System

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
Jet Engine	1		2000	
Fuel Tank	1		25	
Pressure Regulator (Gas)	1		200	
Mount/Gimbal	1		100	
Total			2325	
1. Engine Stabilization				
Bleed Valve Servo Actuator	4	100	400	
Attitude Control Nozzles	4	5	20	
Autopilot System	1	3154	3154	(Item B.2 in this Table)
Total			3564	
2. Gimbal Locking Mechanics*				
Gas Servo Actuators	2	25	50	
Solenoid Valve	1	10	10	
Total			60	
Engine Stabilization and Gimbal Lock (Redundant)			0.21	

* This mechanism is used to lock the engine gimbal with respect to the vehicle in case of failure of the engine stabilization system.

TABLE IX-2 (Cont'd)

B. Autopilot System

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
1. Control and Reference				
Rate Gyro	5	150	750	Reference for engine and vehicle stabilization
Accelerometer	3	100	300	
Engine Throttle Valve S. Actuator	1	120	120	
Vertical Gyro	1	200	200	
Heading Gyro	1	200	200	
Gimbal Potentiometers	2	50	100	
Pilot Control Stick	1	80	80	
Pilot Control Potentiometer and Switch	1	100	100	
Throttle Quad. Potentiometer and Switch	1	100	100	
Connectors and Wiring	-	200	200	
Total			2150	

TABLE IX-2 (Cont'd)

B. Autopilot System (Cont'd)

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
2. Engine Control Electronics				
Transistors	18	15	270	
Transformers	2	50	100	
Resistors, Composition	60	2	120	
Resistors, Precision W.W.	18	4	72	
Capacitors, Paper	18	4	72	
Capacitors, Tantalytic	7	10	70	
Printed Circuit Boards	4	50	200	
Connectors & Wiring		100	100	
Total			1004	

TABLE IX-2 (Cont'd)

B. Autopilot System (Cont'd)

Equipment	No Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
3. Vehicle Control Electronics				
Transistors	21	15	315	
Silicon Diodes	42	8	336	
Resistors, Composition	84	2	168	
Resistors, Precision W.W.	18	4	72	
Capacitors, Paper	24	4	96	
Capacitors, Tantalytic	14	10	140	
Printed Circuit Boards	2	50	100	
Transformer	1	50	50	
Connectors and Wiring	1	100	100	
Total			1377	

TABLE IX-2 (Cont'd)

C. Reaction Control System and Lift Rockets

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
Pressurization and Feed				
N_2 Storage Tank	2	5		
Manual S.O. Valve (N_2)	2	10	0.05	Redundant
Pressure Regulator	2	200		
Check Valve	2	20		
H_2O_2 Tank	2	100	200	
H_2O_2 Relief Valves	2	50	100	
Manual S.O. Valve (H_2O_2)	2	10	-	Redundant and open prior to flight
Rocket Motors Subsystem				
1. Automatic Mode				
Autopilot System	1	1377	1377	Items B.3 in this Table
Manual S.O. Valve (N.O.)	1	10	10	
Solenoid Valves	6	500	3000	
Check Valves	6	100	600	
Thrust Chamber Assembly	8	50	400	
Total			5387	Automatic Mode redundant with Manual Mode

TABLE IX-2 (Cont'd)

C. Reaction Control System and Lift Rockets (Cont'd)

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions Each	Total	Remarks
2. Manual Mode				
Manual S.O. Valve (N.C.)	1	10	10	
Propellant Valves	3	500	1500	
Check Valves	6	100	600	
Thrust Chamber Assembly	8	50	400	
Total			2510	
Redundant Automatic and Manual Rocket Motors Subsystems		Total	14.0	
Lift Rocket Motors				
Manual S.O. Valve	1	10	20	
Propellant Valve	1	200	200	
Check Valves	2	40	80	
Thrust Chamber Assembly	2	50	100	
Subsystem Total			714	

Note: a. No failure rates are shown for fill, vent, and drain valves, since these components are capped prior to flight.

b. It is estimated that each attitude thrust chamber will be pulsed 100 times and each lift rocket will be operated for two minutes during the mission.

c. The failure rate is $314/10^6$ mission, when the system is used for vehicle attitude stabilization alone.

TABLE IX-2 (Cont'd)

D. Electrical System

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
Equipment				
A.C. Generator	1	750	750	
Power Relay	1	50	50	
D.C. Rectifier	1	200	200	
Power Distribution and Control Box	1	300	300	
Connectors and Wiring	10	100	100	
Total			1400	

TABLE IX-2 (Cont'd)

E. Crew Safety Provisions

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions Each	Total	Remarks
Escape Provisions				
"D" Ring Ejection Handle	1	100	100	
Time Delay Cartridges	2	50	-	Redundant
Catapult Rocket	1	300	300	
Drogue Gun Mechanism with Cartridge and Ripcord	1	100	100	
Parachute and Pilot Chute	1	200	200	
Lanyard (Automatic)	1	50	-	Redundant for Seat Bottom Release
Ripcord (Manual)	1	25	-	"
Total			700	
Crew Provisions				
O_2 Tank	1	25	25	
Manual S.O. Valve	1	25	25	
Oxygen Mask	1	50	50	
Protective Suit	1	10	10	
Total			110	

TABLE IX-2 (Cont'd)

F. Vehicle Structure

Equipment	No. Used	Estimated Failure Rate Failures/10^6 Missions		Remarks
		Each	Total	
Structural Legs	4	100	400	
Shock Mounts	4	50	200	
Equipment Mount	-	-	100	
Total			700	

In addition to the redundancies incorporated in the design of the various vehicle systems, provisions have been added for locking the engine gimbal in the vertical position with respect to the vehicle. In case of failure of the electrical or autopilot systems, the vehicle can be safely returned to the ground by locking the engine gimbal and using the reaction control system (automatic or manual mode) for stabilization. Also, the lift rockets are not required for safe descent of the vehicle. Therefore, the total system failure rate for returning the vehicle safely to the ground, once committed to flight, is shown in Table IX-3.

TABLE IX-3

	System	Estimated Failure Rate Failures/10^6 Missions
A	Main Propulsion	2325
B	Engine Stabilization and Gimbal Lock (redundant)	0.21
C	Reaction Control (automatic or manual mode)	314
D	Vehicle Structure	700
	Total	3339

and the reliability of the vehicle for safe descent to ground is: R = 99.66%.

4. Crew Safety Analysis

Another factor of considerable importance in the evaluation of the lunar landing simulator is crew safety. Crew safety is defined as the ability of the system to return the crew uninjured once the mission has been initiated.

From this definition, the dependence of crew safety on system reliability is immediately evident. A 99.66% reliable system as determined from the reliability analysis of the vehicle implies that 9966 times out of 10,000, the crew will be returned uninjured, through successful return of the vehicle to the ground. Thirty-nine times out of 10,000, it will be necessary to return the crew by some auxiliary means. The basic system reliability 99.66% and the probability of failure associated with this value 0.33 immediately establish the need for an auxiliary means to return the crew uninjured. This value further specifies the frequency of use of this auxiliary means at 0.33%, a value equal to the system "unreliability". The auxiliary means of compensating for failures of the vehicle consists of an ejection seat for pilot escape. The mathematical expression which defines crew safety is: $R = 1 - Q_1 \times Q_2$

where Q_1 = probability of a vehicle system malfunction requiring the use of safety provisions.

Q_2 = probability of failure of the safety provisions used.

The product of Q_1 and Q_2 is, therefore, the frequency of occurrence of all possible hazards to the crew. Since the system resulting in the number Q_2 does not operate during a successful mission, it is not a part of the basic vehicle reliability (99.66%). It is, however, of critical importance, as crew safety always take precedence over mission reliability requirements where a conflict exists.

The total failure rate of the crew escape provisions from Table I is $Q_2 = 810/10^6$ missions. Therefore, crew safety is $R_s = 1 - (3339 \times 10^{-6})(810 \times 10^{-6}) = 0.999997$ corresponding to a crew hazard of approximately 3 per million missions.

C. VEHICLE RECOVERY SYSTEMS

In the event of catastrophic failure of the vehicle, pilot safety is provided by a zero altitude zero velocity ejection seat. However, it would be desirable to save the vehicle also, if possible. In order for the vehicle to accomplish an emergency landing under its own power, minimum requirements are that the stabilization and jet engine systems be operating. Redundancy has been provided for vehicle and jet engine stabilization. However, the single engine vehicle presented in this report is in the same class with the French Flying Atar, the British "Flying Bedstead", the Ryan X-13, and the Bell X-14. In all of these, loss of an engine will result in loss of the vehicle. This is characteristic also of the helicopter which, with no forward velocity, is too low for an autorotation recovery below 200 feet. Although sufficient precedent has been established for flying a vehicle which must depend on continued engine operation, it would be desirable to save this vehicle after failure of its engine if the equipment required to accomplish recovery was not so heavy as to compromise the usefulness of the vehicle.

Various methods were investigated for accomplishing vehicle recovery both with and without pilot control. In order to compare the methods, all were based on recovery from 2000 ft altitude and were based on a vehicle weight without the recovery system of 2850 pounds. The weight estimated for each recovery system does not include the growth factor which might be required in the basic vehicle to accommodate the weight of the recovery system. Thus, the total weight penalty for each system will be greater than that indicated, although the weights shown are usable for comparison purpose.

1. Parachutes (Figure IX-1)

In this system, two 130 ft diameter parachutes would be mortar deployed to lower the vehicle to the ground with a terminal velocity of 10.8 ft/sec. The landing gear has been designed for an impact velocity of 10 ft/sec. With this system the pilot could ride the vehicle down or could eject prior to deployment of the chutes. However, once the pilot elected to stay with the vehicle, he could not eject unless he also jettisoned the vehicle chutes, with subsequent loss of the vehicle.

Calculations are based on data from United States Air Force Parachute Handbook, WADC Technical Report 55-265, ASTIA No. AD 118036.

Terminal Velocity:

$$V = \frac{32.7}{D}\sqrt{W}$$

$$V = \frac{32.7}{130}\sqrt{\frac{2850 + 900}{2}} = 10.8 \text{ ft/sec}$$

Weight:

2 - 130 ft diameter chutes of 1.6 oz nylon	830 lbs
2 - mortars	10
2 - containers	60
	900 lbs

Center of Gravity Shift:

3.8 inches upward

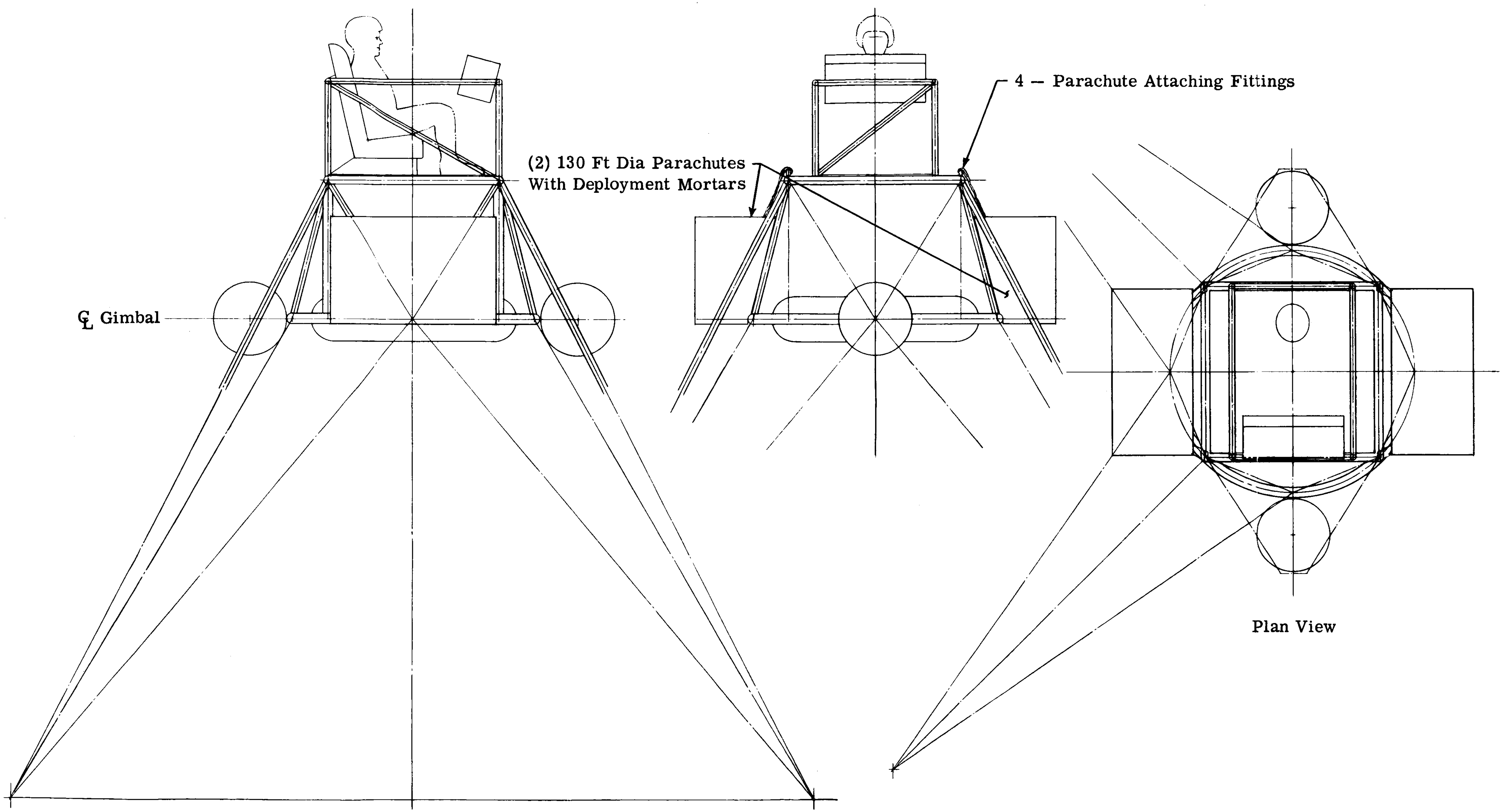


Figure IX-1. Parachute Vehicle Recovery System

2. Flexible Wing (Figure IX-2)

The flexible wing would be stowed in a container, aft of the operator, along with a container of gas at 3000 psi for inflating the keel and leading edges of the wing and a mortar to eject the wing. Shroud lines attached to the simulator structure would position the wing. For a sink rate of 13 ft/sec a wing area of 2800 ft^2 would be required. With this configuration, the pilot must stay with the vehicle in order to control the descent and accomplish a flare at touchdown. Although such systems have been flown successfully, the development of a successful unfurling technique represents a large development problem. In addition, the touchdown forward velocity might be great enough to cause the vehicle to tip over on contact with the ground. This system is considered impractical for this application.

Weight:

1.5 oz Mylar coated nylon	52 lbs
Stowage box, supports, shroud lines, etc.	38
Gas and container	35
Mortar and control	5
	130 lbs

Center of Gravity Shift:

.87 inches upward

3. Rocket Driven Rotor Blades (Figure IX-3)

Rotor blades of 36 ft diameter, driven by solid propellant rockets at the tips would be required for a 10 ft/sec landing of the vehicle. With this system, the pilot may elect to eject or stay with the vehicle. However, if he elects to stay with the vehicle, once the rotor is turning, he cannot change his decision.

Weight:

Rotor blades - 2 at 100 lbs each	200 lbs
Solid prop. rocket motors - 2 at 15 lbs each	30
Upper structure support - tubing	100
- hub and bearings	50
	380 lbs

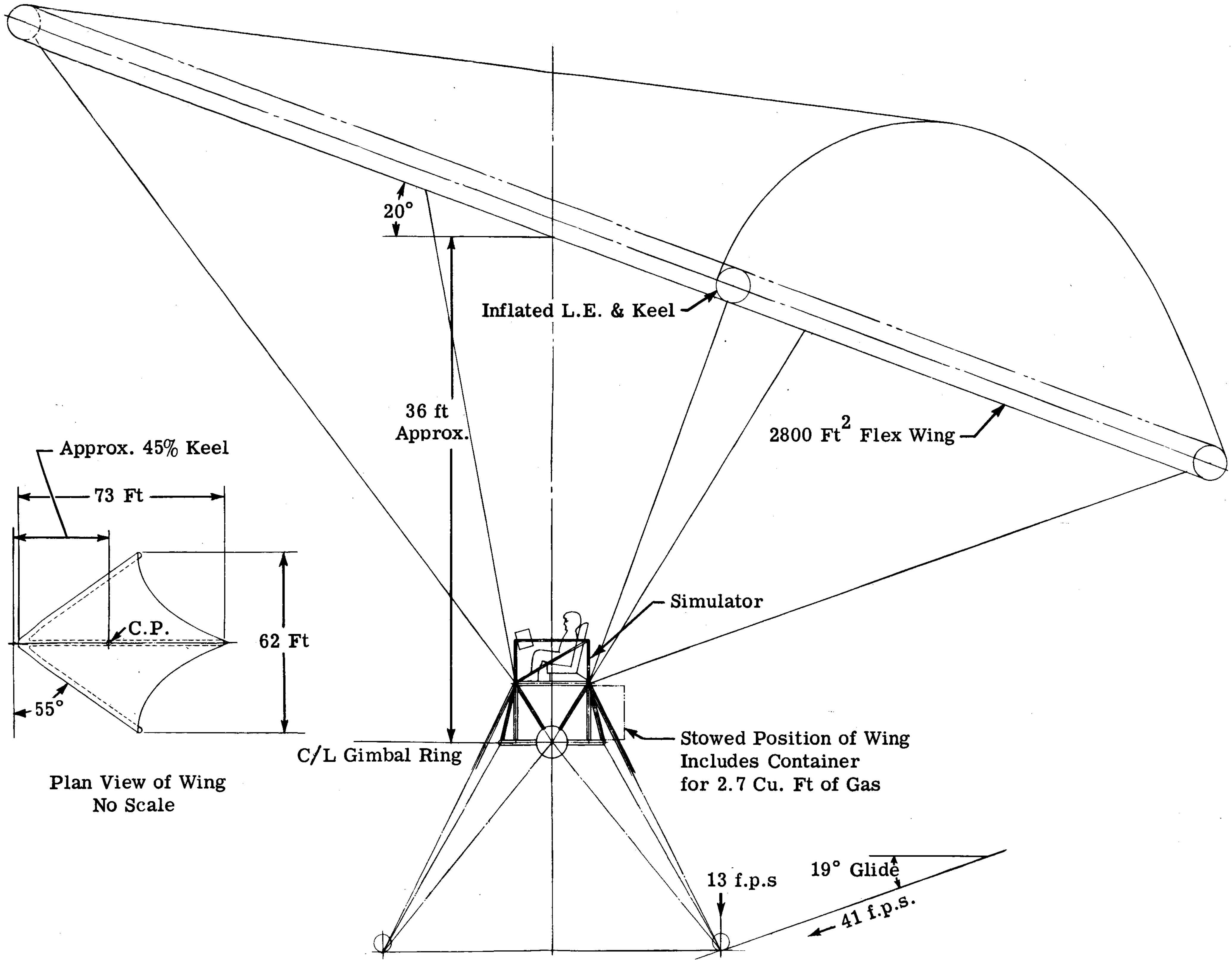


Figure IX-2. Flexible Wing Vehicle Recovery System

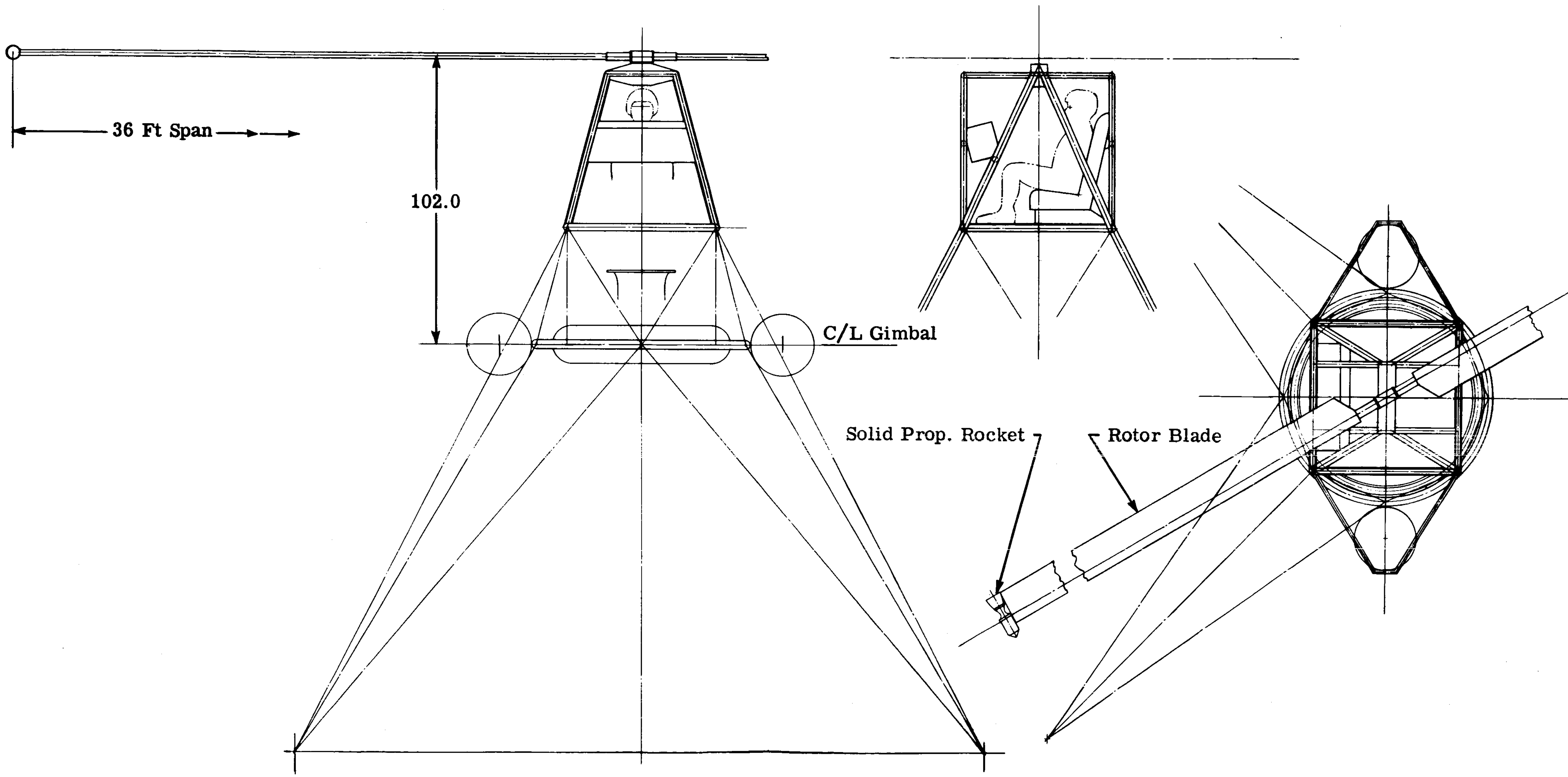


Figure IX-3. Rotor Blade Vehicle Recovery System

Center of Gravity Shift:

11.2 inches upward

The weight of this configuration, the large shift in center of gravity which it causes, and the high drag which it will impose on the vehicle all dictate against incorporation of the rotor blade recovery system.

4. Multiple Jet Engines

Vehicle reliability could be enhanced by a multi-engine configuration. In a two-engine configuration, the thrust vector from each engine operating alone must pass through the vehicle center of gravity, accomplished by mounting the jet engines at an angle from the vertical or mounting the engines horizontally opposed into a common thrust diverter. Either configuration results in a much larger, heavier vehicle requiring higher thrust engines. An alternate would be a four-engine cluster around the center of gravity with the engines operating in pairs. Either pair would have a resultant vector through the center of gravity capable of lifting the vehicle.

The multi-engine configuration was rejected because it departs too far from the concept of an early availability low cost vehicle. Therefore, no configuration drawing was made for this concept. However, the concept has been considered for a future larger three-man vehicle.

5. Lift Rockets

Since rocket engines present a very favorable thrust to weight ratio, propellant load was calculated for a recovery system based on the employment of two rocket engines mounted on the vehicle airframe. The assumption is made that descent from 2000 ft and landing can be accomplished in 60 seconds. Rocket hardware is calculated to weigh 70 pounds. For liquid bipropellants with a specific impulse of 240 seconds, 860 pounds of propellant would be required. For hydrogen-peroxide with a specific impulse of 120 seconds, 2000 pounds of propellant would be required. Since these weights are prohibitive, no layout was made of this configuration.

With this concept, the pilot must stay with the vehicle to control the landing rockets.

6. Rockets plus Parachute, Manual Control (Figure IX-4)

The vehicle is lowered with a 55 ft diameter parachute at a terminal velocity of 33.4 ft/sec. This velocity is reduced to zero at touchdown by employing two throttleable 3000 pound peroxide retro rockets controlled by the pilot. One second of full thrust of these rockets is sufficient. However, in order to eliminate critical timing of the initiation of the retro rocket, propellant has been provided for 2-1/2 seconds burning time. This is also sufficient to save the vehicle in a free fall from 200 ft, the altitude below which the chute could not be deployed in time. This system has the advantage that the equipment can be located to cause minimum shift of vehicle center of gravity. The pilot must remain with the vehicle to control the retro rockets.

Terminal Velocity:

$$V = \frac{32.7}{D} \sqrt{W}$$

$$V = \frac{32.7}{55} \sqrt{2850 + 304} = 33.4 \text{ ft/sec}$$

Weight:

2 - 3000 thrust chambers at 35 lbs	70 lbs
2 - valves at 6 lbs	12
Mounts, throttle control, plumbing	20
H_2O_2 tank	12
H_2O_2	117
N_2	5
	236 lbs
55 ft diameter chute	53
Chute container, supports, anchor fittings, etc.	10
Mortar and control	5
	304 lbs

Center of Gravity Shift:

.4 inches upward

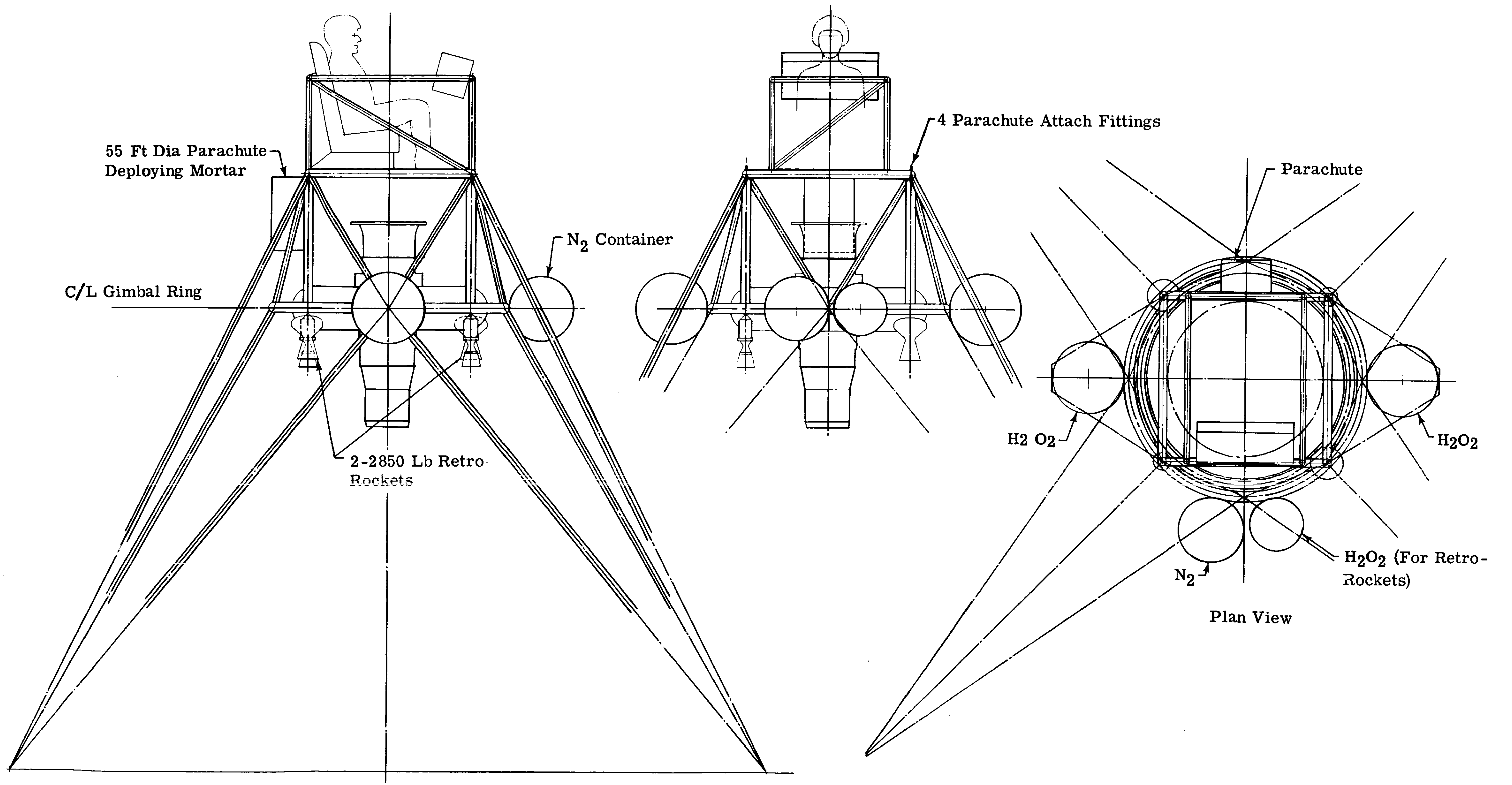


Figure IX-4. Parachute and Retrorocket Vehicle Recovery System

7. Retro Rockets plus Parachute, Automatic Control

This concept is similar to that presented in the paragraph above and shown in Figure IX-4, except that the peroxide rockets are replaced with solid propellant rockets. The solid propellant rocket firing is initiated by means of an altitude sensing trigger. A trade-off is possible between parachute weight and retro rocket weight. As parachute size is decreased, its weight goes down and the weight of the retro rocket required goes up. However, the compromise cannot be based on weight alone, but must consider the type of altitude trigger used. As parachute size decreases, sink rate increases and the retro rocket must be fired at a higher altitude. If a mechanical type trigger such as a bob weight is to be employed, this would limit the altitude for firing a retro rocket to about 32 ft. However, if a radar altimeter were employed, then recovery could be initiated at any altitude. A second consideration involves a minimum altitude from which successful chute deployment can be obtained. If this minimum altitude is 200 ft, then the retro rocket must be capable of absorbing the energy of a 200 ft drop. Retro rocket thrust has been set at 3-g. Table IX-4 shows the important parameters for various chute sizes.

TABLE IX-4

Parachute diameter, ft	55	28	24	18
Parachute weight, lbs	58	18	15	11
Retro Rocket Weight, lbs (T = 8700, I_{sp} = 240)	23	41	47	59
Trigger Weight, lbs	10	10	10	10
System Weight, lbs	91	69	72	80
Terminal Velocity, ft/sec	32	64	75	96
Rocket initiation altitude, ft	8	32	44	72
Free fall recovery altitude, ft	24	96	132	216

It can be seen that the weight penalty is not prohibitive with this arrangement. In addition, the system is fully automatic and can be effective whether the pilot stays with the vehicle or not. However, it does depend on the development of a successful retro rocket trigger mechanism. It is recommended that this system be considered for further work and incorporation in the vehicle when a successful trigger mechanism is perfected. Since the components of this system can be mounted in almost any location on the vehicle, the basic vehicle design need not be compromised and the balance of the vehicle will not be materially affected.

Table IX-5 presents a summary of the seven vehicle recovery systems considered. None have been incorporated in the vehicle design presented because of the high reliability estimated for the CF-700-2B engine and the weight penalty involved. However, System No. 7 merits further study to evaluate accurate altitude sensing devices and to decrease weight.

TABLE IX-5

Recovery System	Weight (Lbs)	C.G. Travel (Inches)	Remarks
1. Parachutes	900	3.8	With or without pilot.
2. Flexible Wing	130	.87	Pilot control required tip-over at landing development problem.
3. Rotor Blades	380	11.2	Pilot control required high drag.
4. Multiple Jet Engine	na	na	Larger vehicle required.
5. Lift Rockets	930	1.0	Pilot control required.
6. Rocket and chute, manual	304	.4	Pilot control required.
7. Rocket and chute, automatic	91	1.0	With or without pilot requires development of alt. sensor

X. AEROSPACE GROUND EQUIPMENT REQUIREMENTS

Figure X-1 illustrates the operational flow and activities anticipated for the lunar landing vehicle program. The aerospace ground equipment proposed for support of this program is based upon this flow sequence and consists of the following items:

A. TESTING, MEASURING AND ADJUSTING

1. Electronic Multimeter (GFE)

A standard laboratory quality volt-ohmeter such as the Hewlett-Packard Type HP-410B. This meter is considered adequate for monitoring of the flight control system, for polarity checks, static sensitivity checks, static gain checks, and system frequency response indications. Accessible test points and connecting leads will be provided for adapting this unit to the vehicle systems.

2. Ejection Seat Test Set (CFE)

This unit will be procured from the seat manufacturer and used to verify functional continuity of the ejection circuitry.

B. HOISTING, JACKING, LIFTING AND WEIGHING

1. Engine Sling Set (CFE)

The jet engine will be installed and removed using standard chain falls and a sling set. This sling will consist of braided steel cables with spreader bars and fittings for attachment to the engine lift eyes and to the chain fall hooks.

2. Vehicle Sling Set (CFE)

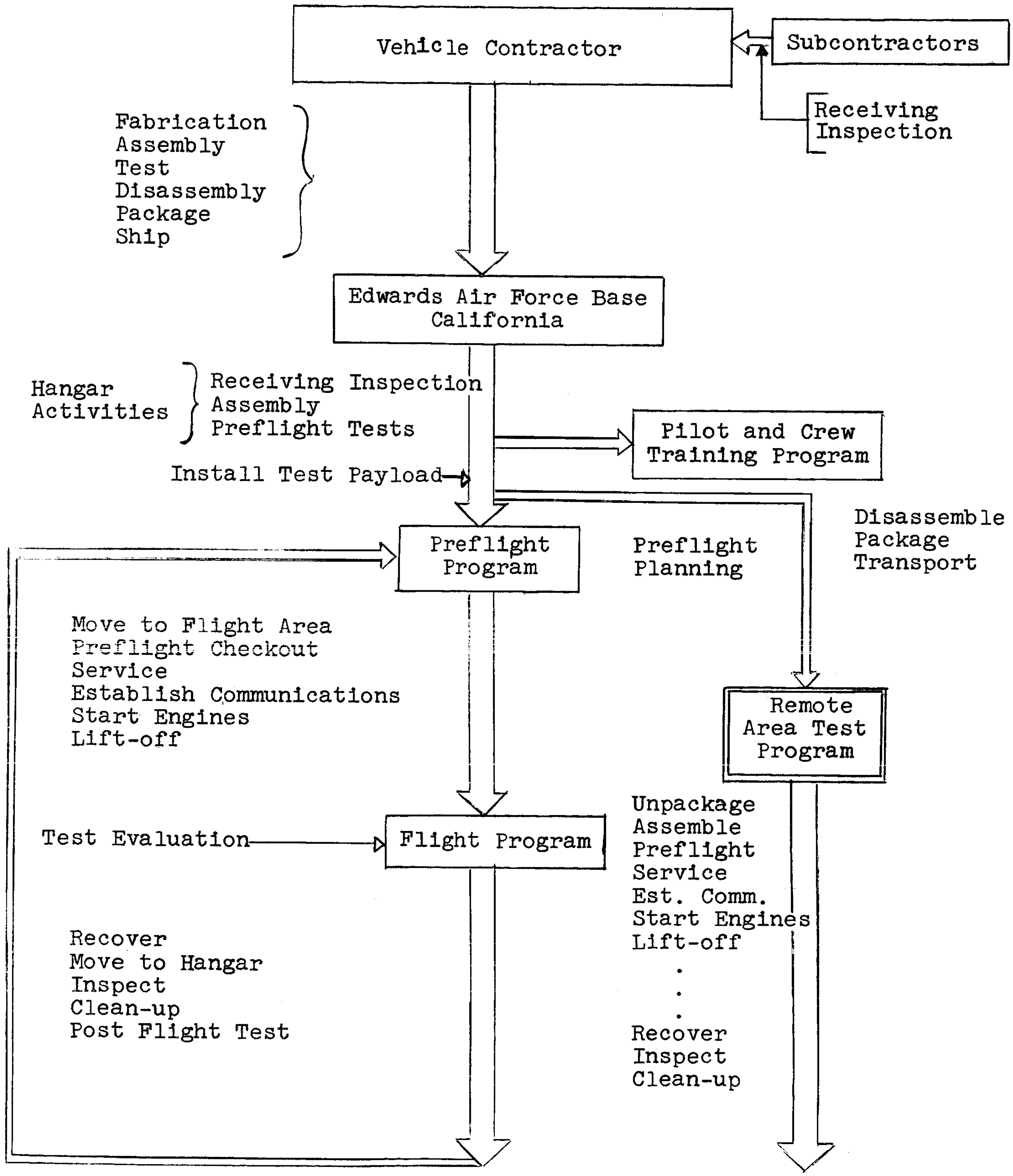
A requirement exists for lifting of the complete lunar landing vehicle during assembly and test operations. A braided steel cable and spreader yoke assembly will be designed to permit lifting the vehicle with an overhead hoist or crane.

3. Chain Falls (GFE)

Handling of the simulator jet engine and other large components will require standard manually operated chain falls (reference Item B.1) of 3000 lbs capacity.

FIGURE X-1

SYSTEM FLOW AND ACTIVITIES CHART



4. Truck Mounted Crane (GFE)

Figure X-1 indicates that flying at a remote area may be scheduled as a part of the program. Thus it becomes necessary to remove and replace the vehicle legs while in the field. It may also be necessary to retrieve the vehicle from an area remote to the base. A standard truck mounted crane will answer this requirement. This crane should have a boom extension and load capacity sufficient to permit lifting of the vehicle using the vehicle sling set (reference Item B.2). Some commercially available cranes meeting this requirements are:

a. Insley Type K, Lorry Crane with a 40' boom having a capacity of 3700 lbs at this radius without outriggers.

b. P and H Model 255A-TC Truck Crane with a 40' boom having a capacity of 4250 lbs at this radius without outriggers.

The Mobile Aircraft Handling Crane furnished by AFFTC for the X-15 program might also be utilized.

5. Center of Gravity Fixture (CFE)

Measurement of the vehicle center of gravity and functional testing of the H_2O_2 propulsion system will require that a device be provided which can support the vehicle using lift pads on the jet engine mount. Once in an elevated position, the vehicle attitude can be observed and center of gravity corrections made as required. Similarly, the H_2O_2 jets can be activated and vehicle attitude noted as qualitative indications of reaction control performance.

A tubular frame fixture having lift pads which mate with the jet engine mounting structure and sufficiently high to permit adaption to a standard forklift will meet this requirement. Elevation of vehicle will be accomplished by alignment of this fixture beneath the engine gimbal, emplacing a forklift truck in the fixture lift fittings and elevating the entire assembly to the required height.

6. Forklift Truck (GFE)

A commercial type forklift truck having a capacity of 4000 lbs and a vertical lifting range of six feet is required for use in determining the vehicle center of gravity and for H_2O_2 system functional testing (reference Item B.5).

7. Weighing Kit (GFE)

A standard weighing kit type D-1 four load type or its equivalent will be needed for weight measurements of the vehicle.

C. POWER GENERATING AND STARTING

1. Power Unit D.C. (GFE)

An external source of 28 volt D.C. electrical power will be required for ground test and checkout of the vehicle. Units in inventory which will meet this requirement are many and varied. They include the MA-2 or the MD-3A multipurpose servicing units (self-contained mobile) or the B-8 portable power supply, which requires a 220/440 volt 60∿ 3 Ø source of input power.

2. Power Unit 400 Cycle A.C. (GFE)

An external source of 400 cycle 3 Ø electric power will be required for ground test and checkout of the vehicle. The MA-2 or MD-3A multipurpose servicing units noted in Item C.1 will also deliver the 400 cycle power required to meet this requirement.

3. Air Compressor (GFE)

An external source of compressed air and connecting hoses will be required for starting the jet engine. Air requirements consist of 100 lbs/min @ 43 psia at a temperature of 360°F. The type MA-1A, or the type MC-1 (modified) air compressors will meet the performance requirements.

D. PROPELLANT SUPPLY

1. Gaseous Nitrogen Supply (GFE)

The vehicle propellant system requires 2.37 cu ft of nitrogen gas at a pressure of 3000 psig. Delivery of N_2 gas to the vehicle tank can be accomplished using the type MD-1, MD-3, or other available compressed gas cylinder trailers.

2. Hydrogen Peroxide Supply (GFE)

The rocket system has a tank capacity of 800 lbs (67 gallons) of 90% hydrogen peroxide. It is recommended that the X-15 aircraft servicing trailer (E5201) be employed. The X-15 trailer has sufficient capacity (75 gallons), has an N_2 pressure transfer system and a supply of water for flushing and washdown. An alternate servicing method is direct gravity transfer of H_2O_2 from the shipping drums to the vehicle tanks. Drum handling devices and an H_2O_2 filter would be required to support this operation.

3. Adapter Hoses and Fittings (CFE)

These hoses and fittings are required to adapt the X-15 peroxide trailer for servicing the vehicle rocket propulsion system. The hoses will be of the flexible type constructed of a material which is compatible with 90% H_2O_2 and sufficiently long to reach from ground level to the vehicle tank level. End fittings will be of the quick disconnect variety to simplify the H_2O_2 servicing task.

4. JP-4 Supply (GFE)

The vehicle jet engine uses standard aircraft JP-4 fuel. The JP-4 tank capacity is 500 lbs (77 gallons). Filling of this tank can be accomplished using a standard jet aircraft fuel servicing trailer such as the type F6, type M131A-2 or the type A-1B.

5. Engine Oil Supply (GFE)

The jet engine uses lubricating oil conforming to MIL-L-7808C. Approximately four quarts are required to fill the tank. Standard gallon containers of this oil poured directly into the engine tank when needed will meet this requirement adequately.

6. Pilot's Air Supply (GFE)

The tank will require filling with compressed contaminate-free breathing air at 2250 psi.

E. PROTECTIVE EQUIPMENT

1. Protective Cover(s) (CFE)

Cotton duck covers will be provided for the pilot control section and engine. These covers will serve to protect critical components from the effects of environment during storage and transport.

2. Protective Clothing (GFE)

Personnel involved in handling the hydrogen peroxide propellant must wear protective clothing. In general, Grayolite suits, hoods and rubber boots will be adequate for this purpose.

F. MAINTENANCE EQUIPMENT

1. Maintenance Platform(s) (GFE)

Accessibility must be provided so that field personnel can accomplish the various servicing and maintenance tasks required for operation of the vehicle. Personnel maintenance platforms having a working elevation of at least 16 feet will be needed. The type B-2 or B-3 adjustable aircraft maintenance platforms will meet this requirement.

G. TRANSPORTING AND TOWING

Figure X-1 indicates that two major moves will be required in the flight research program, first,from the contractor factory to Edwards Air Force Base in California and secondly, from Edwards Air Force Base to the remote test area and return. The assembled vehicle is approximately 20 feet high by 21 feet wide by 21 feet long. By removing the four legs, the seat, the jet engine and portions of the pilot's protective structure, a package approximately 8 feet x 7 feet x 8 feet can be realized. The removed legs will form four packages, each about 16.5 x 4 x 5.5 feet. These packages, as well as those containing the jet engine and the pilot's seat, fall within the limitations listed for highway, rail and air transportability in the 80.5 handbook. The following items of equipment will be required to implement this basic transportation scheme:

1. Center Body Transport Skid (CFE)

This skid will be designed to support the vehicle center body structure and components during the major transporting events. It will be fabricated from welded commercial steel channels and equipped with forklift, hoisting and tiedown fittings, and mounting pads which mate with ceter body strong points.

2. Ejection Seat Shipping Container (CFE)

3. Jet Engine Shipping Container (CFE)

It is planned that the containers in which the seat and jet engine are shipped to Bell Aerosystems Company will be retained and used without modification for additional transportation of these items.

4. Low Bed Semitrailer (GFE)

5. Truck, Tractor (GFE)

Items No. 4 and 5 will be needed at Edwards Air Force Base for movement of the vehicle subassemblies to and from the remote test area. Semitrailers conforming to type XM-269 and tractors type M48 or their equivalents can be used for this purpose.

6. Vehicle Handling Casters (CFE)

During flight testing, a means must be provided for short distance movement of the vehicle. Four, easily removable castered wheel assemblies will be provided for this purpose. These units will consist of 360° full castering, hard rubber wheels having foot-operated brakes and attach to the landing struts with easily operated clamps. Attachment of these casters can be accomplished by relieving the cylinder pressure on one strut at a time to a point where the units can be clamped into place, then repressurizing the strut.

H. SERVICING EQUIPMENT

1. Tool Kit (GFE)

Only standard tools will be required at Edwards Air Force Base for support of the flight program. No special tools are required for assembly, test or checkout.

I. COMMUNICATIONS EQUIPMENT

1. Transmitting-Receiving Set (GFE)

Standard VHF communications equipment will be required. This is available at the Edwards test site. Considerations must also be given, however, to the communications requirements while at the remote site.

J. MISCELLANEOUS

1. Jet Impingement Plate (CFE)

During takeoff, the vehicle jet engine will produce a temperature of approximately 960° at ground level. Prolonged operation at this temperature will damage and cause spalling and melting of concrete or asphalt aprons and runways. A jet impingement plate will be provided for use during captive portions of the vehicle program wherein extended ground operations over hard surfaces are anticipated. This plate will be attached to the surface of the runway or apron and be sufficiently large to cover the jet impingement area. A .250" thick aluminum plate six feet square would fulfill this requirement.

XI. WIND TUNNEL AND FLIGHT RESEARCH FACILITY REQUIREMENTS

A. WIND TUNNEL PROGRAM

Drag and drag moments have been calculated by analytical methods described in Section IV of this report. Because of the simplifying assumptions required in the analysis of this truss type construction, it is recommended that more accurate data be obtained by wind tunnel testing.

It should be noted on the schedule (Figure XII-1) that test data will not be available in time to change the basic airframe configuration. However, the data will be useful for making final design adjustments of the configuration, for completing control system studies, and will form the basis for calculating final vehicle performance, flight trajectory data and performance limits.

It is recommended that wind tunnel tests be conducted on a powered model of approximately 1/5 scale in the NASA Langley 300-mph, 7 x 10 foot wind tunnel. The model will be tested at full-scale Reynold's number while the engine mass flow parameter is simulated, provided it is possible to obtain sufficient simulated engine mass flow to correspond to the high velocity required for full-scale Reynold's number. It will be necessary to measure both the simulated propulsion system forces and moments and the aerodynamic loads on the model airframe, because the full-scale propulsion system will be gimbal-mounted to the airframe with its own stabilizing system. It will be necessary to instrument the model propulsion system supports to determine the propulsion system forces and moments so they can be subtracted from the total forces and moments measured by the tunnel balance.

Measurements in all six degrees of freedom will be obtained from the tunnel balance. Tests should be made through an angle-of-attack range of -180 to +180 degrees, if possible, and through a sidestep range of 45 degrees. To obtain full-scale maximum Reynold's number will require testing at approximately 240 miles per hour. The propulsion system will be designed so that it can be positioned with respect to the airframe in varying attitudes up to 45 degrees from the airframe axis. The propulsion system mass flow will be simulated by an air-driven fan or electric-motor powered propeller. The turbojet inlet and aft fan inlet will be simulated, and the proportion of air flow through each will be determined by proper sizing of the turbojet and aft fan ducts.

B. FLIGHT RESEARCH FACILITY AND PERSONNEL

The major portion of this report has delineated the requirements and established the feasibility and characteristics of a free flight vehicle which can simulate lunar landing on earth. The analysis has established the need and the practicality of producing such a vehicle in the immediate future. However, for the vehicle to be a practical research tool, supporting facilities and personnel must be available. This section will briefly outline the facilities and personnel recommended for optimum utilization of this vehicle.

The vehicle should be flown at a test site that gives the best visual simulation of the lunar environment. Some of the visual factors desirable are high brightness, glare, and contrast, and a light colored surface devoid of vegetation. In addition, the atmosphere should be clear, with long distance visibility. The area should have a high percentage of cloudless days. These conditions will give a good approximation of the visual environment expected near the lunar surface.

Other sections of this report list specific items of support equipment required to handle and service the free flight lunar landing simulator. However, additional general supporting services are required to obtain maximum usefulness from the vehicle.

Standard air base facilities for vehicle storage and handling are required. This includes hangar space with overhead door clearance of 20 ft. Also required are standard forklifts, cranes, tugs, and flat bed trailers, with personnel to operate them.

Facilities and personnel are required for servicing the vehicle with hydrogen peroxide and jet fuel. Facilities for servicing the X-15 and personnel trained in the use of hydrogen peroxide are adequate for the lunar landing vehicle. Standard jet aircraft fuel servicing equipment is suitable. In addition, personnel experienced in jet engine maintenance should be available.

To obtain research data, ground tracking equipment is required to provide vehicle trajectory data. Telemetry is required to collect vehicle data for both research and flight safety. It will also be desirable to obtain data on operator motor performance through the use of bioastronautic instrument packages. Data processing equipment should be available in order to make rapid use of information collected in simulated missions. VHF aircraft radio facilities will be required for communication from ground monitoring and control stations, and tracking stations to the vehicle.

Experienced test pilots should be available at the facility using this vehicle. Analog computer facilities are required for preflight training of these pilots, and to test trajectories and control methods prior to using them in flight. Helicopters and helicopter pilots are desirable to fly chase for flight safety and to obtain in-flight photographic coverage.

XII. SCHEDULES

The time required from receipt of order to delivery of one flight research test vehicle was estimated using Program Evaluation Review Techniques (PERT). The PERT chart will be found in the envelope inside the back cover of this report. Elapsed time between events was established by the groups which will accomplish the work. Lead time on vendor items was obtained from the vendors. These time estimates were entered in the PERT IBM program and the longest or most critical path determined. This path is shown by the heavy arrows on the PERT diagram. As shown, the vehicle will be out of the shop, ready for initial flight test in ten months, and available for delivery to the customer in 11-1/2 months.

The program plan shown in Figure XII-1 is constructed from the PERT analysis. The longest path leads through the development of the hydrogen peroxide lift rocket and attitude control system. This path is 2-1/2 weeks longer than the next most critical path. If this path were shortened by 2-1/2 weeks, three other paths then become critical. The first of these is caused by jet engine delivery time, eight months being quoted by General Electric Company. The present schedule is made possible by building the vehicle with a mockup engine. The actual engine will be installed after the start of composite systems testing as indicated in Figure XII-1. The next critical path is the electronic flight control system, and the next the time required for the landing gear shock absorber to be designed, fabricated and tested by the vendor.

This schedule is believed to be realistic and can be met, based on the following:

1. It is assumed that the basic vehicle design and basic subsystem designs established in the Phase I study are accepted.

2. Contractor flight testing will be limited to a demonstration of the controllability and maneuverability of the vehicle for a reasonable spectrum of speeds and altitudes.

3. This schedule is based on a 40-hour week, one shift operation. The total program time can be shortened by an extended work week on the part of the prime contractor. In this case, however, vendor schedules must also be reduced accordingly.

Figure XII-2 shows the availability and research use of the free flight lunar landing flight research vehicle in relation to the Apollo program. The Apollo major milestones were obtained from the NASA Project Apollo Statement of Work, dated December 18, 1961, Part 1, Figure 1. Since the manned control center, controls, displays, and attitude controls which will be used in a lunar landing are located in the command and service modules, these designs must be defined before the lunar landing module has to be committed to production. It is urgent, therefore, that flight research be accomplished at an early date. Figure XII-2 indicates the urgency of starting the research program in time for its results to be incorporated in the Apollo design.

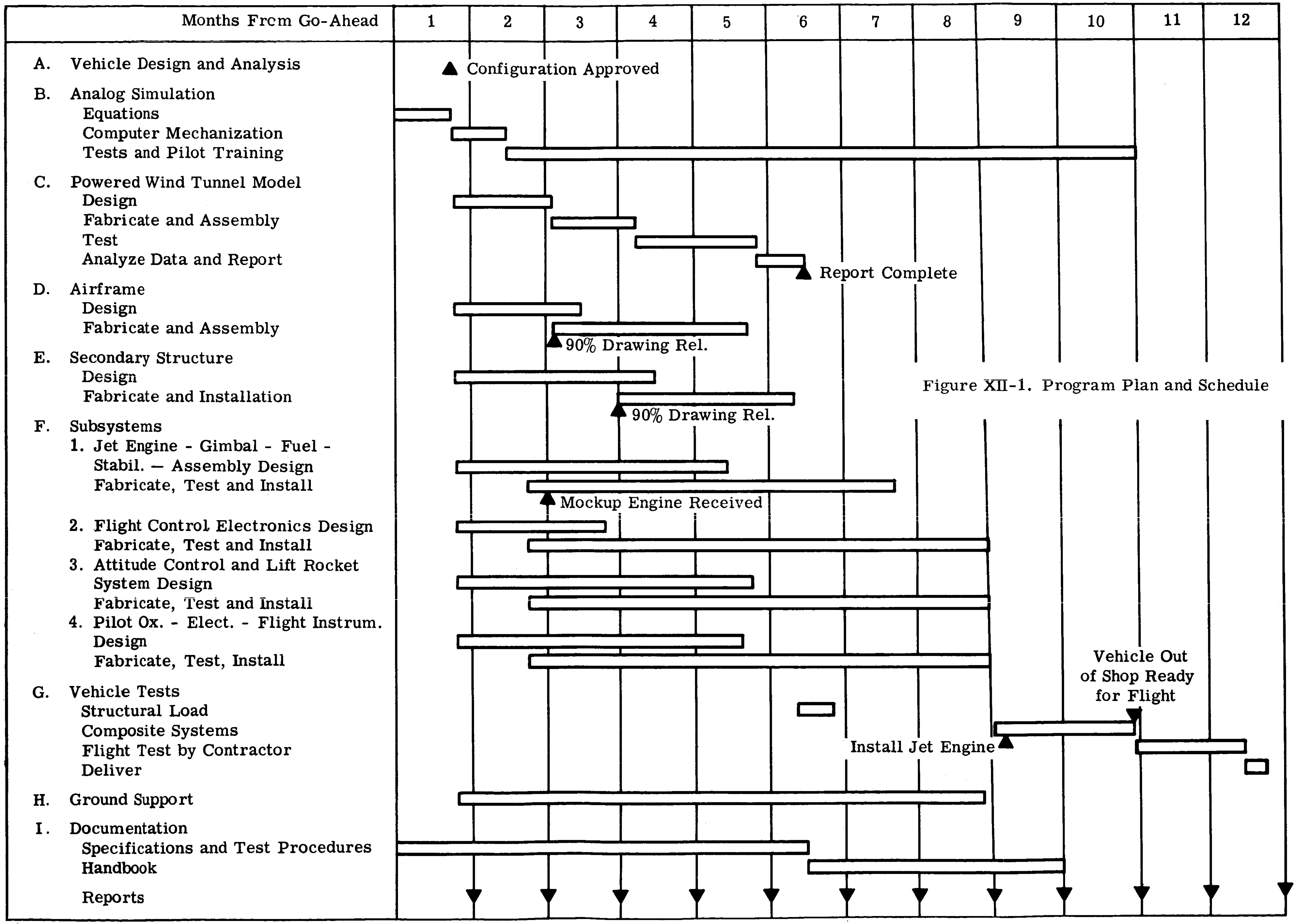


Figure XII-1. Program Plan and Schedule

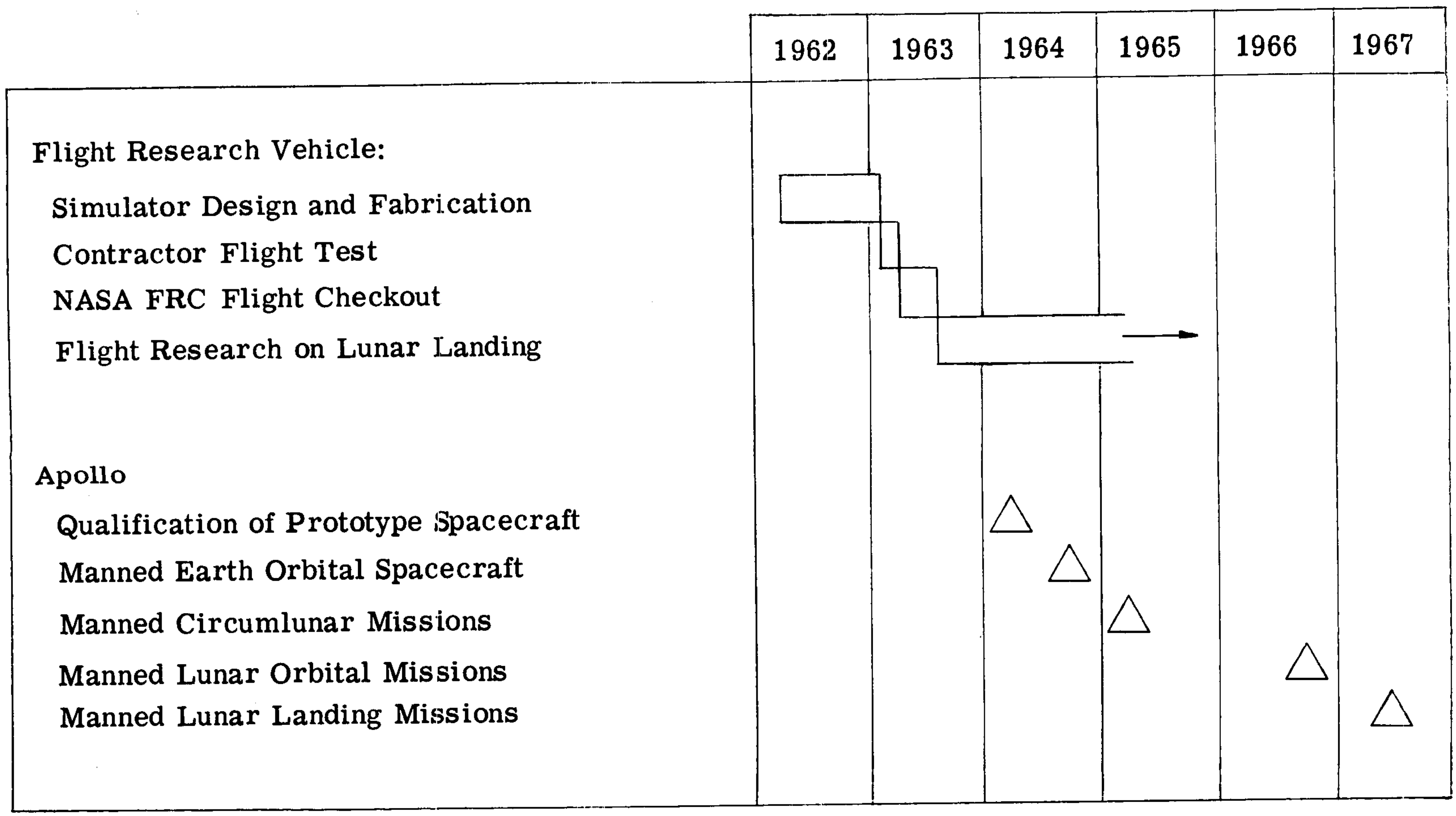


Figure XII-2. Research Vehicle and Apollo Schedule

XIII. COST ANALYSIS

The cost to design, manufacture, and test a one-man vehicle, as presented in Figure III-1 is estimated as follows:

Vehicle:

Engineering	$690,000
Manufacture	550,000
Component and System Test	420,000
Jet Engine	150,000
Contractor Flight Test	136,000
	$1,946,000
Wind Tunnel Program	54,000
Ground Support Equipment	38,000
Handbooks and Field Support	34,000
Fifty-hour Airworthiness Test on Jet Engine, Vertical Operation	140,000

This estimate is based on 1962 prices, using a maximum of already developed hardware, as discussed in other sections of this report.

XIV. FUTURE GROWTH

A. BIPROPELLANT ROCKET SYSTEM

A hydrogen peroxide lift rocket and attitude control system has been selected because it can meet the vehicle requirements at lower cost and with earlier delivery than can a bipropellant system. However, bipropellant systems are under development for space applications, which can be adapted for this vehicle. A discussion of a bipropellant system is given in Section VI-E of this report.

The bipropellant system can replace the peroxide system and result in a substantial weight saving. This can be used to increase payload, or to increase operating time of the lift rockets, so as simulate several lunar landings between refuelings.

B. ACCESSORIES FOR ADDITIONAL RESEARCH

The vehicle design presented in this report provides a maximum of flight research versatility consistent with minimum vehicle weight, cost, and delivery time. An effort has been made to incorporate in the basic vehicle structure the capability to perform as many as possible of the research tasks which can be anticipated at the present time. It has been necessary to eliminate some capability to save weight. However, these tasks and missions can be accomplished by simple modifications to the basic vehicle. The additional tasks might be accomplished by field modification kits described in the following:

1. Adjustable Leg Angle

The angle which the legs make with the vertical has been selected to give a vehicle tipover angle adequate for a safe landing on earth. However, in the future, it may be desirable to experiment with landing gear configurations which provide increased or decreased tipover stability. A modification kit would provide four new legs capable of angular adjustment with respect to the airframe. These legs would be identical to the present legs except that the upper section of the outer longeron would be of adjustable length and pin connected at both ends. Adjustable fittings attaching the shock absorbers to the legs would compensate for the change in angle when the landing legs are adjusted.

2. Three-Legged Configuration

The proposed vehicle is equipped with four legs because this results in minimum weight, and provides a convenient mounting point for reaction controls. However, it may be desired to compare four-legged configurations with three-legged configurations. This can be accomplished by a field installed leg adapter ring. The four legs are removed from the vehicle, and the adapter ring attached to the present leg attachment points. Three of the present legs are then attached to the adapter ring. The adapter ring includes outriggers on which the present reaction controls would be mounted, so that the same control system can be utilized with the three-legged configuration. This modification is shown in Figure XIV-1.

3. Gimballed Lift Rockets

Vehicle attitude control is accomplished in the proposed vehicle by means of 16 fixed reaction jets. However, it may be desirable to accomplish research on attitude control by means of gimballed lift rockets. This can be accomplished by a field modification kit which involves remounting the present two lift rockets on gimbals. The gimballed rockets would be mechanically linked to the pitch, roll, and yaw pilot controls. Flexible peroxide lines would be provided to the rockets.

4. Rocket for Lateral Translation

In the present vehicle, lateral translation is accomplished by tilting the vehicle so that the lift rockets provide a horizontal component of thrust. It may be desired to accomplish research on a control system in which the vehicle remains vertical with lateral translation accomplished by means of a fixed mounted horizontal rocket. This would be accomplished by attaching to the present vehicle, one additional thrust chamber identical to those used as lift rockets. It would be mounted horizontally in the plane of the center of gravity, facing rearward. It would be provided with a separate throttle control. Peroxide would be supplied from the presently installed tanks, which have reserve capacity for 200 additional lbs of peroxide.

5. Vehicle Leveling Device

The proposed vehicle will rest at the same angle as the terrain on which it lands. It may be desirable to accomplish research on a landing system which allows the vehicle to remain in a vertical position in spite of landing on hilly or sloping ground. This may be accomplished by replacing the present shock absorbers with a new combination shock absorber/load leveling device. This consists of a long stroke hydraulic piston and cylinder on each leg. The four cylinders are manifolded to allow oil to flow between cylinders. As the legs contact the ground, they will adjust themselves to uneven terrain until the last leg touches down. At this point, energy would be absorbed by compression of a standard shock absorber connected to the hydraulic circuit. After the vehicle has come to rest, valves to each cylinder would be closed, thus hydraulically locking the leveling devices so as to hold the vehicle in a vertical position, with all four legs firmly in contact with the ground.

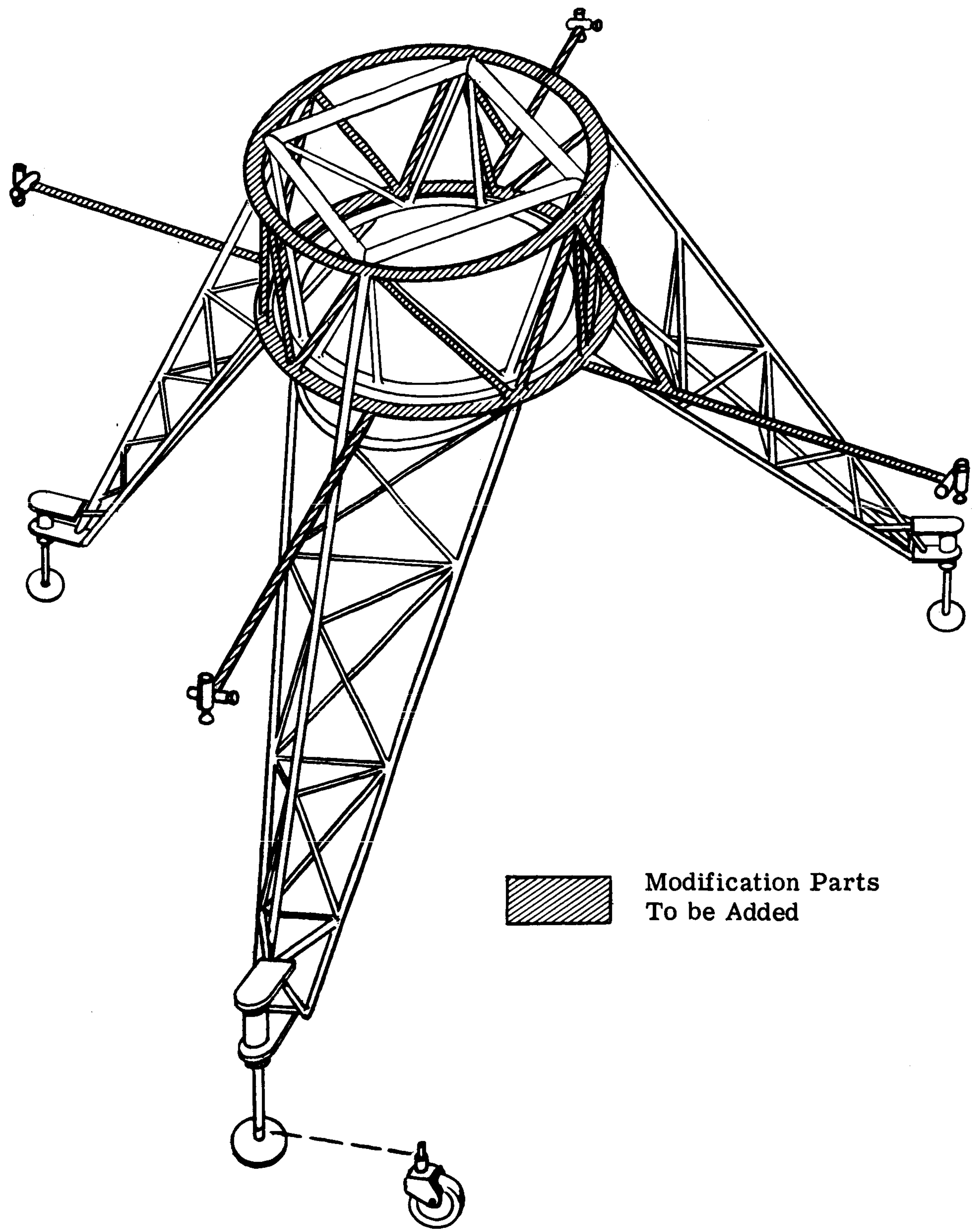


Figure XIV-1. Three-Legged Configuration Adapter Kit

XV. THREE-MAN LANDING VEHICLE SIMULATOR

The present study has concentrated on the investigation and design of a small, one-man lunar landing simulator. This vehicle will satisfy the early needs of NASA to conduct basic free flight research and systems development testing of the problems associated with lunar landing. Provision has been made in the one-man vehicle design to substitute an additional man for other payload, so that early investigation can be made of multicrew operations.

As the development of a lunar landing vehicle proceeds, however, the need may exist for more extensive earth based free flight testing with a larger crew,and utilizing more complete assemblies of actual lunar flight systems and subsystems. A typical test which can be envisioned would involve the use of a simulated Apollo Command Module. Investigation could be made of crew position and function, ground visibility, displays and controls, and flight control systems. For such tests, much larger payload capability would be required than can presently be obtained with the single engine one-man vehicle. Payloads for a multimanned vehicle in the range of 3000 to 6000 lbs are envisioned.

A preliminary design concept of such a vehicle has been developed. Essentially it is a scaled-up version of the one-man vehicle, employing the same basic design principles and features. In developing this design, consideration was given to both single engine and multi-engine configurations. Tentatively, the multi-engine configuration has been selected because of greater payload resulting from the superior thrust to weight ratio of the smaller engines, and increased safety since the vehicle can be designed to have engine out capability.

The three-man landing vehicle simulator as shown in Figure XV-1 is a multi-engine, four-legged open truss-work structure supporting an Apollo spacecraft type capsule. Its overall dimensions are 25 ft by 25 ft and 39 ft high. With a propellant load permitting 20 minutes of turbojet operation and 8 minutes of main lift rocket operation, the vehicle gross weight is approximately 17,000 lbs. Four basic units comprise this vehicle, a platform structure, landing legs (4), a gimballed engine mount, and the capsule.

Two aluminum alloy tubular rings 12 feet in diameter and separated seven feet by tubular truss-work forms the all-welded platform structure. Fittings on the upper ring secure the capsule to the vehicle. Landing leg fittings are integral with both rings, and gimbal axle housings are provided diametrically opposed in the lower ring. The gimbal ring is a steel tube containing four sets of gimbal bearings spaced radially at 90°.

Each landing leg is fabricated by welding aluminum alloy tubing to form a tapered, triangular truss. Pin joint fittings are provided at the upper ends of the three longerons of each leg for attachment to the platform structure and to permit removal during shipment. Landing shock struts are fastened to the lower extremity of each leg.

A tubular steel engine mount of welded construction contains two gimbal axle housings diametrically opposed for attachment to the gimbal ring. Eight General Electric SJ-132 turbojet engines rated at 3050 lbs thrust at sea level on a standard day are supported vertically in the mount. The engines surround a cylindrical JP-4 fuel tank which is also supported by the engine mount. Jet stabilization autopilot system components are also installed on this mount.

Two throttleable main lift rocket engines producing 1500 lbs (max) of thrust each are mounted to the underside of the upper horizontal strut between the lower longerons on two opposing landing legs. Sixteen reaction control rockets, clustered in sets of four, are mounted on the lower end of each landing leg. H_2O_2 propellant for all the rocket systems is stored in four spherical tanks trunnion mounted to the platform lower ring between the landing legs. Two high pressure N_2 spheres for propellant tank pressurization are trunnion mounted to the lower ring or the platform structure within two of the landing leg trusses.

The capsule is an off-loaded replica of the actual Apollo spacecraft. Life support systems, equipment, heat shields, etc., not required for the simulated flight are eliminated. Controls, instrumentation and crew support equipment required for simulator operation are installed in the capsule.

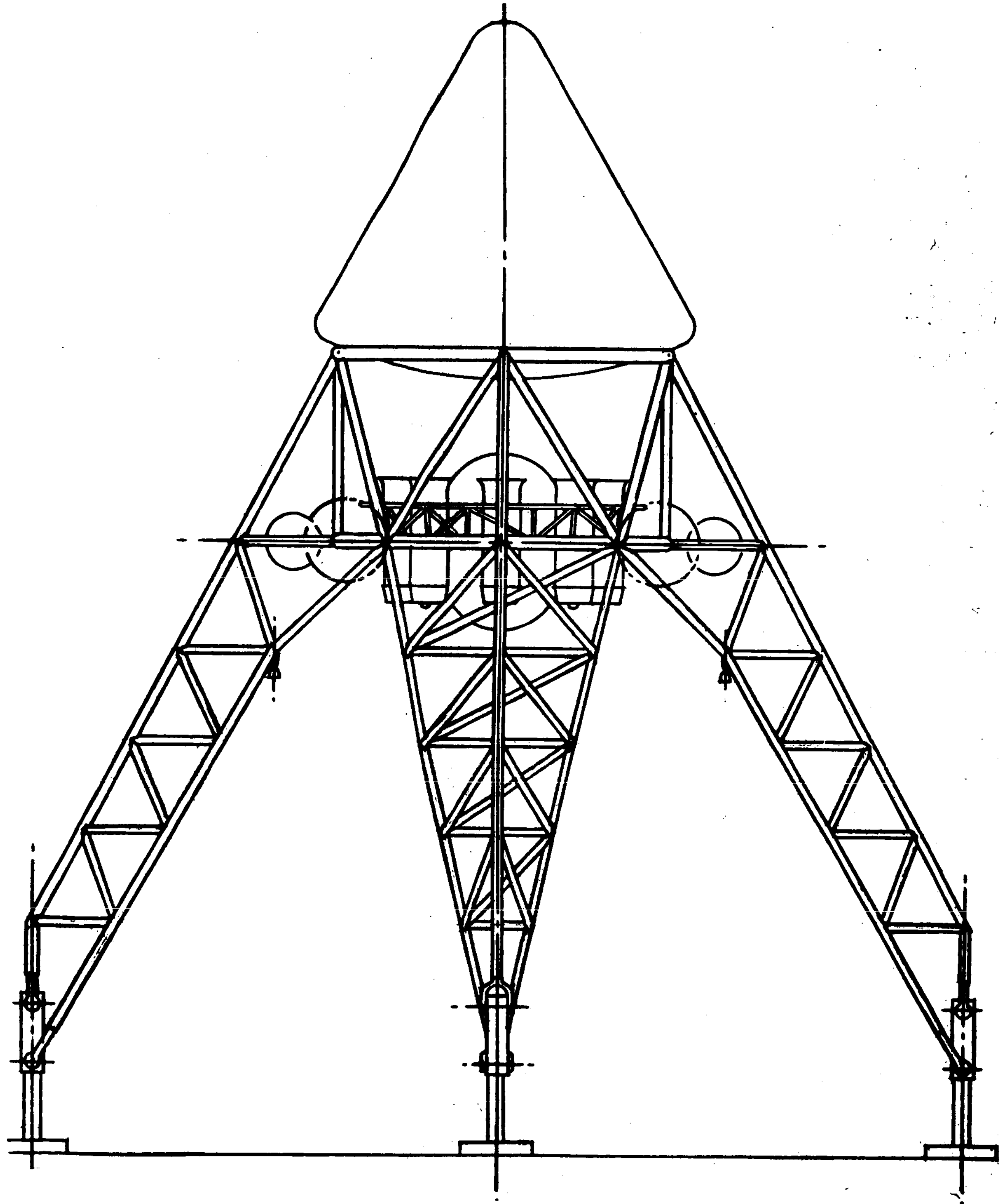
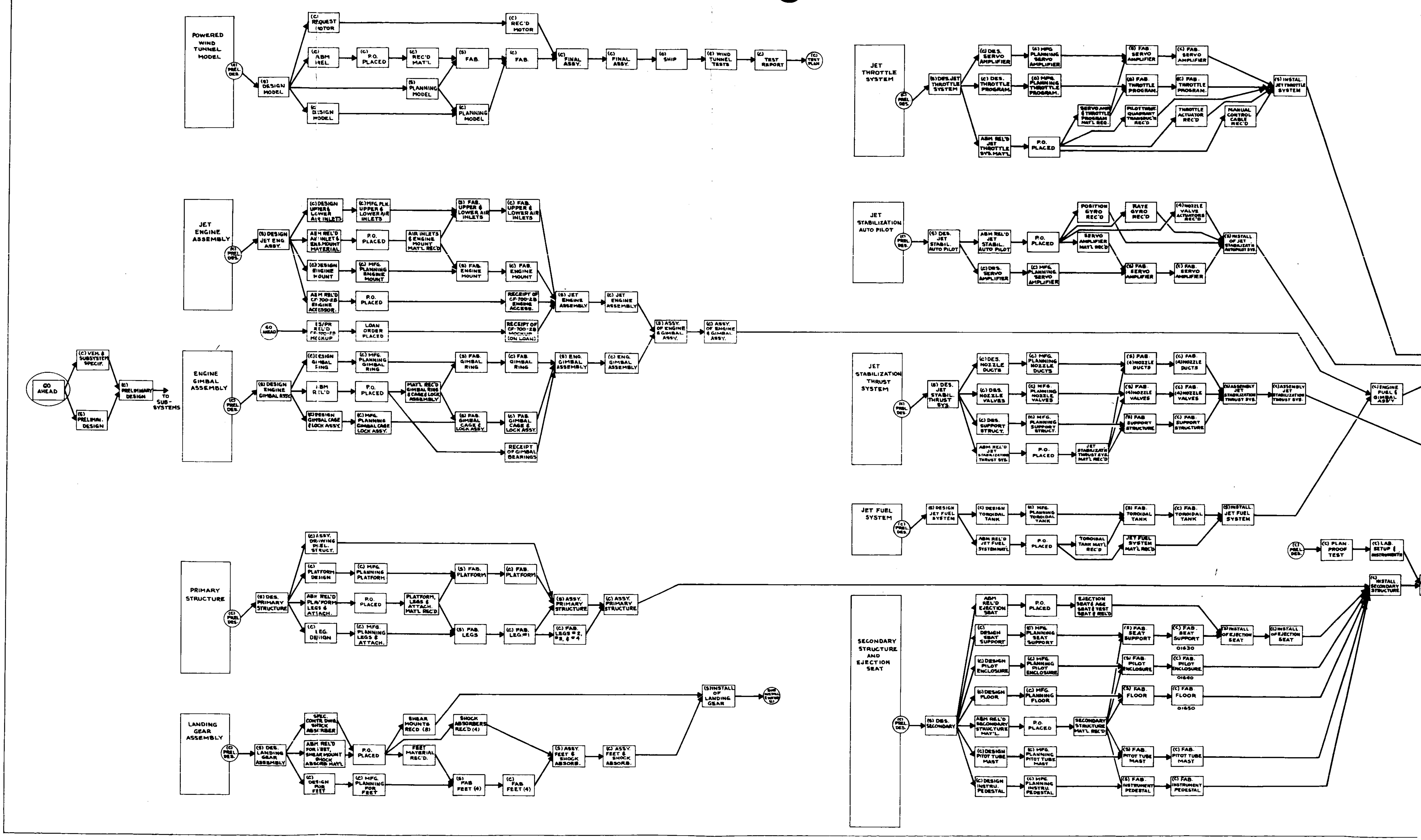
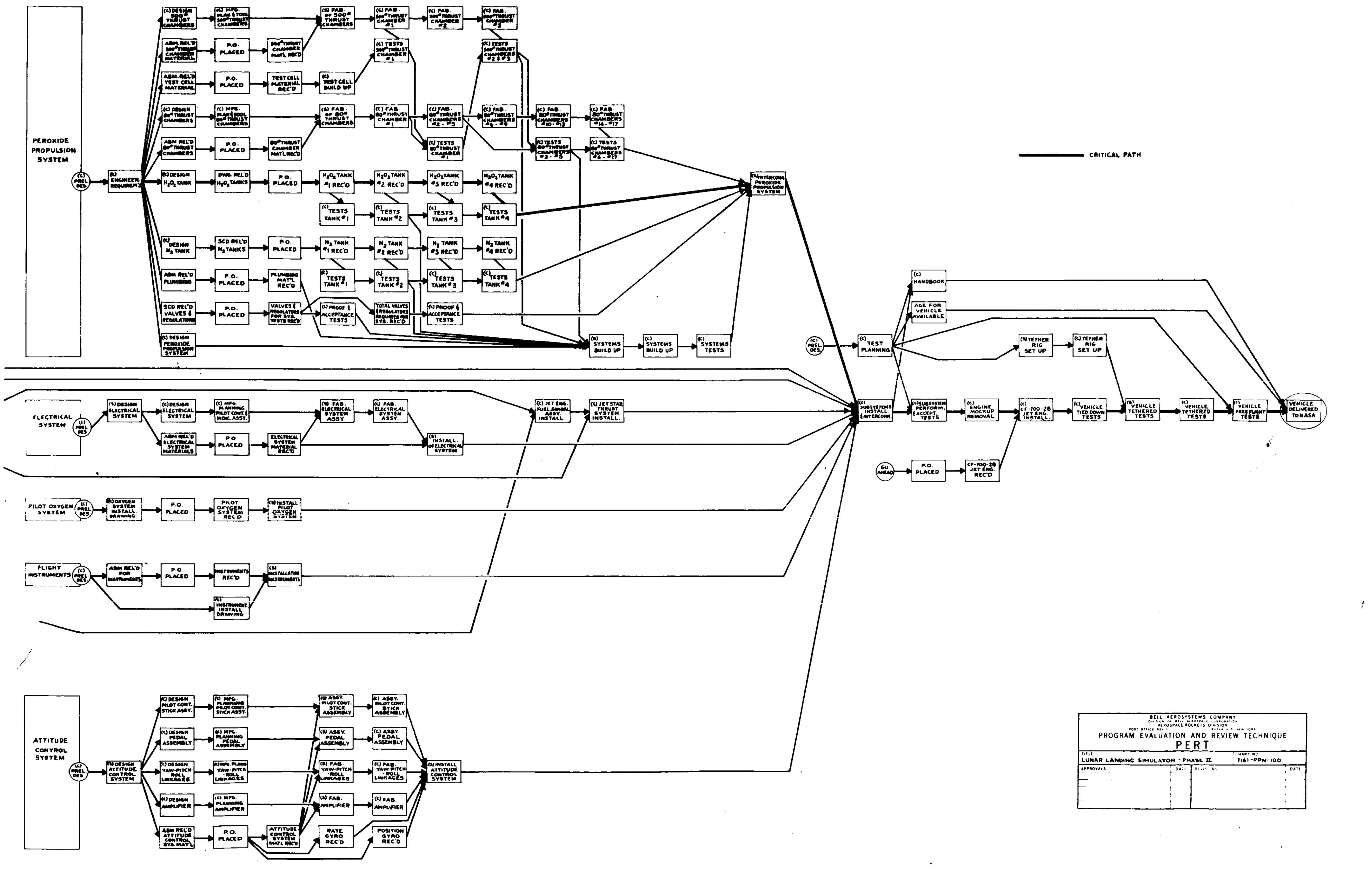


Figure XV-1. Three-Man Lunar Landing Simulator


POWERED WIND TUNNEL MODEL
JET ENGINE ASSEMBLY
ENGINE GIMBAL ASSEMBLY
PRIMARY STRUCTURE
LANDING GEAR ASSEMBLY
JET THROTTLE SYSTEM
JET STABILIZATION AUTO PILOT
JET STABILIZATION THRUST SYSTEM
JET FUEL SYSTEM
SECONDARY STRUCTURE AND EJECTION SEAT
GO AHEAD
TO SUB-SYSTEMS


PEROXIDE PROPULSION SYSTEM
ELECTRICAL SYSTEM
PILOT OXYGEN SYSTEM
FLIGHT INSTRUMENTS
ATTITUDE CONTROL SYSTEM
CRITICAL PATH
TEST PLANNING
HANDBOOK
AGE FOR VEHICLE AVAILABLE
SUBSYSTEMS INSTALL. & INTERCONN.
ENGINE MOCKUP REMOVAL
CF-700-2B JET ENG. INSTALL.
VEHICLE TIED DOWN TESTS
VEHICLE TETHERED TESTS
VEHICLE FREE FLIGHT TESTS
VEHICLE DELIVERED TO NASA
GO AHEAD
P.O. PLACED
CF-700-2B JET ENG. REC'D
BELL AEROSYSTEMS COMPANY
AEROSPACE ROCKETS DIVISION
PROGRAM EVALUATION AND REVIEW TECHNIQUE
PERT
TITLE
LUNAR LANDING SIMULATOR - PHASE II
CHART NO
7161-PPN-100
APPROVALS
DATE
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